4. SPACECRAFT OPERATIONS

This section provides information regarding spacecraft subsystem operations, with a special emphasis on those areas requiring real-time monitoring, trend and performance analysis, routine ground intervention, or possibly special interaction in the event of a spacecraft anomaly. A background description of subsystem components and operations is also provided.

4.1 COMMAND AND DATA HANDLING

The Command and Data Handling (C&DH) subsystem is comprised of hardware and software elements performing services of command validation and distribution, telemetry processing, data storage, stored command processing, clock maintenance and time distribution, and health and safety. Hardware elements of the C&DH subsystem are referred to as the Spacecraft Data System (SDS) and software tasks as the Flight Data System (FDS). The majority of operations performed by the FOT are accomplished through FDS provided services. The ACS task resides on the ACS processor of the SDS and is discussed separately in Section 4.2.

The primary operations with which the FOT will interact with the C&DH subsystem include table and memory operations, stored command processing, time maintenance, and recorder management. Before continuing with descriptions of these operations, a description of the SDS architecture and the common spacecraft and ACS operating system are in order. The FDS is based on a layered architecture, with lower levels accessing hardware and performing device driver functions, and higher levels performing FDS functions, as well as providing an interface to the other subsystems and the ground.

4.1.1 Spacecraft Data Systems

The TRMM design includes two SDSs, providing full redundancy, and are referred to as side A and side B. Each SDS contains two 80386 microprocessor cards (S/C and ACS Processors), an uplink card, a downlink card, a clock card, 14 memory cards, and four Low Voltage Power Converters (LVPCs). The two SDS units are physically located in separate boxes on the spacecraft. However, cross strapping amongst most components between the two SDSs is supported. This means a complete fail over to the redundant unit is not always required in the event of a single component failure. Figure 4.1-1 provides a functional and interface diagram of the SDS.

To support mission objectives, certain components of the backup SDS (side B) will be powered ON during normal mission operations. The following paragraphs describe the hardware components of the SDS. Table 4.1-1 summarizes nominal on-orbit configuration and allowable cross-strapping between side A and side B.

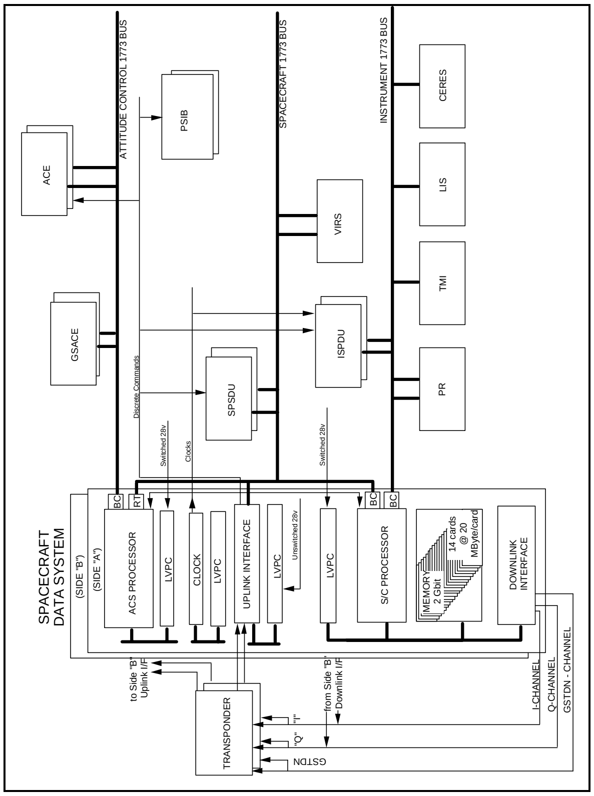


Figure 4.1-1 TRMM Spacecraft Data System

Component	Side A Status	Side B Status	Cross-Strap Capability
ACS Processor	ON	OFF	Yes
S/C Processor	ON	OFF	Yes
Memory Cards (14)	ON	OFF	No
Downlink Card	ON	OFF	Via alternate card
S/C Clock Card	ON	OFF	Yes [1]
Frequency Standard	ON	OFF	No
S/C Uplink Card	ON	ON	Yes

Table 4.1-1 SDS On-Orbit Component Configuration

Note:
[1] The Clock cards provide redundancy in such a way that the 1 Hz pulse from each Clock card is provided to both IPSDUs (A and B), and Time packets can be provided to either S/C Processor. However, the Clock Cards do not provide redundancy with respect to the FS. FS A is configured to Clock card A, and FS B is configured to Clock card B.

The SDS supports processing of a number of Special or Hardware commands. Special commands allow for C&DH reconfiguration by ground commands. Special commands provide for spacecraft subsystem reconfiguration by the Uplink Card, independent of FDS software or the 1773 data bus. Special-Special commands refer to those processed by the Uplink card directly. These commands are discussed individually in the following sections. Table 4.1-2 summarizes special command functions.

Special Command Type	User	Description
+5 V	C&DH	• Reset processor • Change BC/RT Status
+28 V	C&DH PSIB	• Power ON/OFF S/C and ACS processors • Power OFF units saturating S/C bus ("blabbermouth mode").
Special-Special	Uplink Card	• Switches to other uplink card. • Disable output of uplink card in "blabbermouth mode."
Software-Special		

Table 4.1-2 Special Commands

4.1.1.1 386 Microprocessor Boards

Two 386 Microprocessor boards provide the platform to support FDS and ACS software. The microprocessor boards are designated as the ACS and S/C Processors. The ACS Processor acts as the Bus Controller for the ACS bus, while the S/C Processor acts as BC for the S/C and Instrument busses.

Each board contains an Intel 80386 16 MHz microprocessor, an 80387 math coprocessor, an 82380 Direct Memory Access (DMA) controller, as well as associated application specific integrated circuits. Each board contains 448 KB Electrically Erasable Programmable Read Only Memory (EEPROM), of which 64 KB is allocated as Boot Strap PROM, 128 KB allocated as Boot PROM, and 256 KB as PROM. Only the 256 KB PROM is programmable in flight. Each board also contains 1 MB Harris Static Random Access Memory (SRAM). RAM is radiation hardened to provide protection against Single Event Upsets (SEUs).

All of TRMM's flight code and default operating parameters are stored in EEPROM. At system boot, code is copied into RAM, from which it executes. Due to changing conditions in flight, the FOT may be required to re-load EEPROM. Operationally, this works the same way as other table and memory loads and dumps, which are covered in Section 4.1.5. Typically, operations will center on RAM, where stored commands and other operations data are stored. Figure 4.1-2 illustrates how these memory areas are segmented and arranged on the S/C Processor card and to which functions they are dedicated.

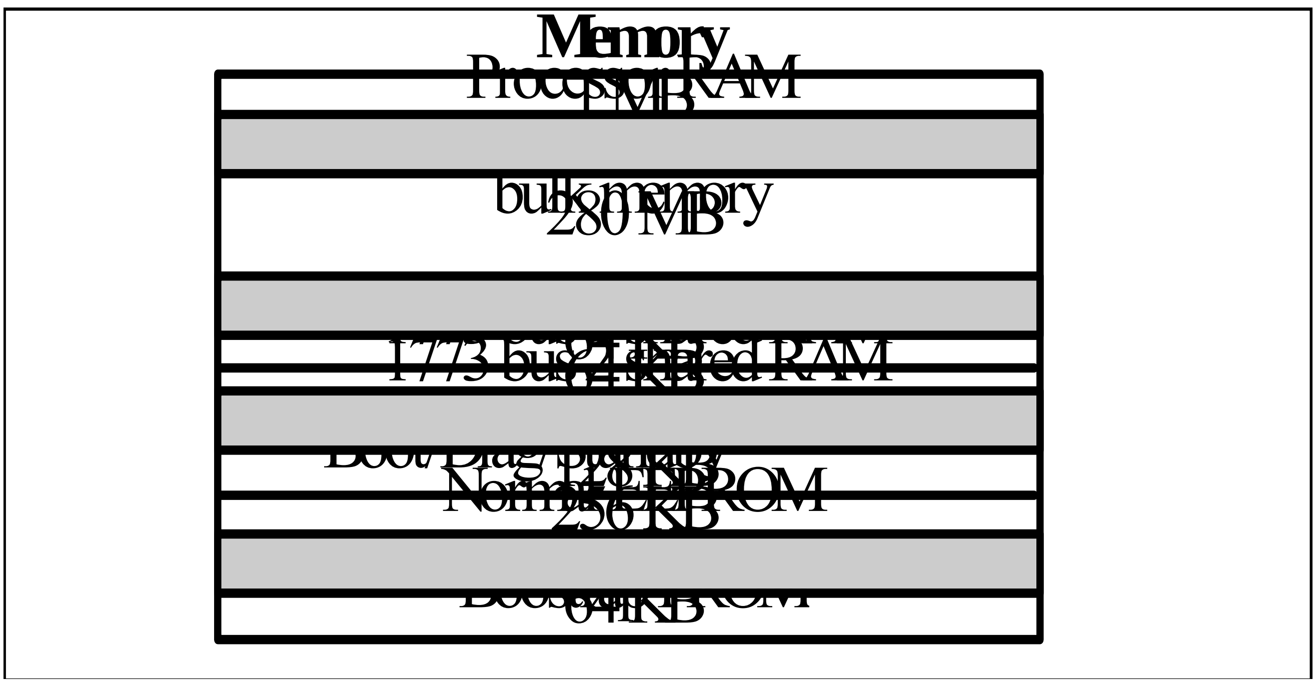


Figure 4.1-2 Spacecraft Memory Map

4.1.1.2 Bulk Memory Cards

Bulk memory provides approximately two orbits worth of storage for recorded housekeeping, science, and event message data. Bulk memory consists of 14 memory cards that contain 25 MB total storage per card. Each card has 20 MBytes of addressable memory available for data storage and an additional 40 MBits of check bits. In addition, 5 MBytes is used for Error Detection and Correction (EDAC) storage. Separate bulk memory cards exist on side A and side B, however, there is no access to bulk memory from the alternate SDS side.

Approximately 150 Single Event Upsets (SEUs) are expected in the 280 MB of bulk memory per day. Typically, however, not all of memory will have data recorded in it. All memory will be checked for errors through EDAC scrubbing once per orbit. EDAC will correct SEUs and detect and report multiple bit-flips.

Bulk memory is equipped with a Latch Protection circuit to eliminate the possibility of encountering a Single Event Latchup (SEL) during memory access. An SEL occurs when a part hangs up, draws excessive current, and/or will no longer operate until the power to the device is cycled OFF and ON. The latch protection circuit operates off a +5 Volt standby power supply. Figure 4.1-3 details the latch protection circuit.

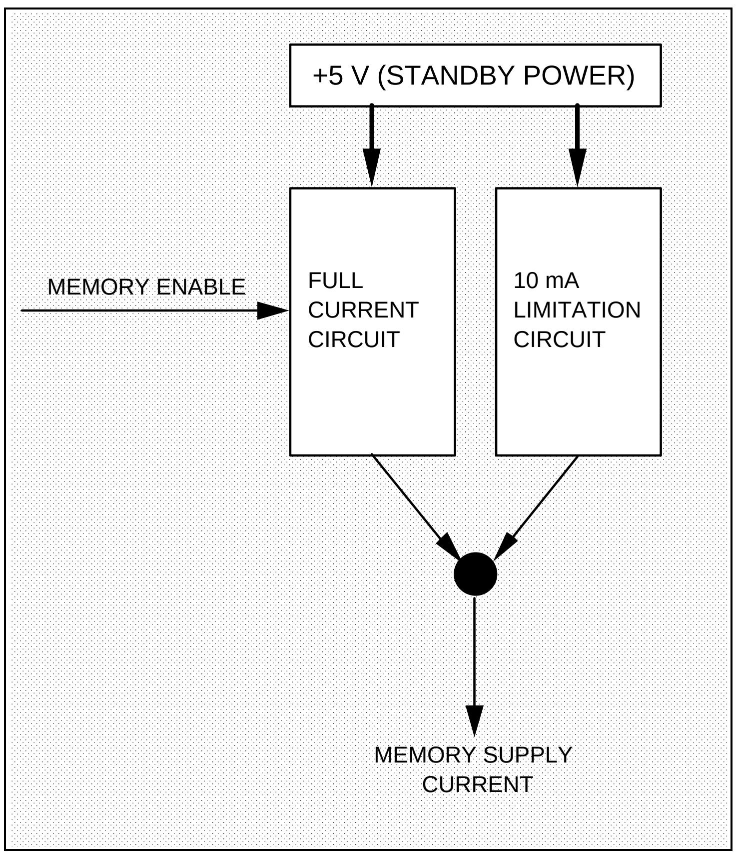


Figure 4.1-3 Latch Protection Circuit

4.1.1.3 Uplink Card

The Uplink card provides the interface between the RF Communications subsystem and the S/C Processor for the receipt of command data. Command data from both transponder receivers are passed to both Uplink cards. Both Uplink cards interpret the data and perform receiver selection between the two channels. The Uplink card receives command data, detects the Command Link Transmission Unit (CLTU) Start Sequence, verifies code block parity, and passes code block data to the S/C Processor.

The S/C Processor receives data from both Uplink cards, however, only one Uplink card is designated as active at a time. Commands received from the active Uplink card are distributed and executed. Commands from the inactive card are monitored for the Change Uplink Card command, which may be received on either channel. The FDS maintains the CLCW for both active and inactive Uplink cards. Figure 4.1-4 demonstrates the nominal configuration of the Uplink cards.

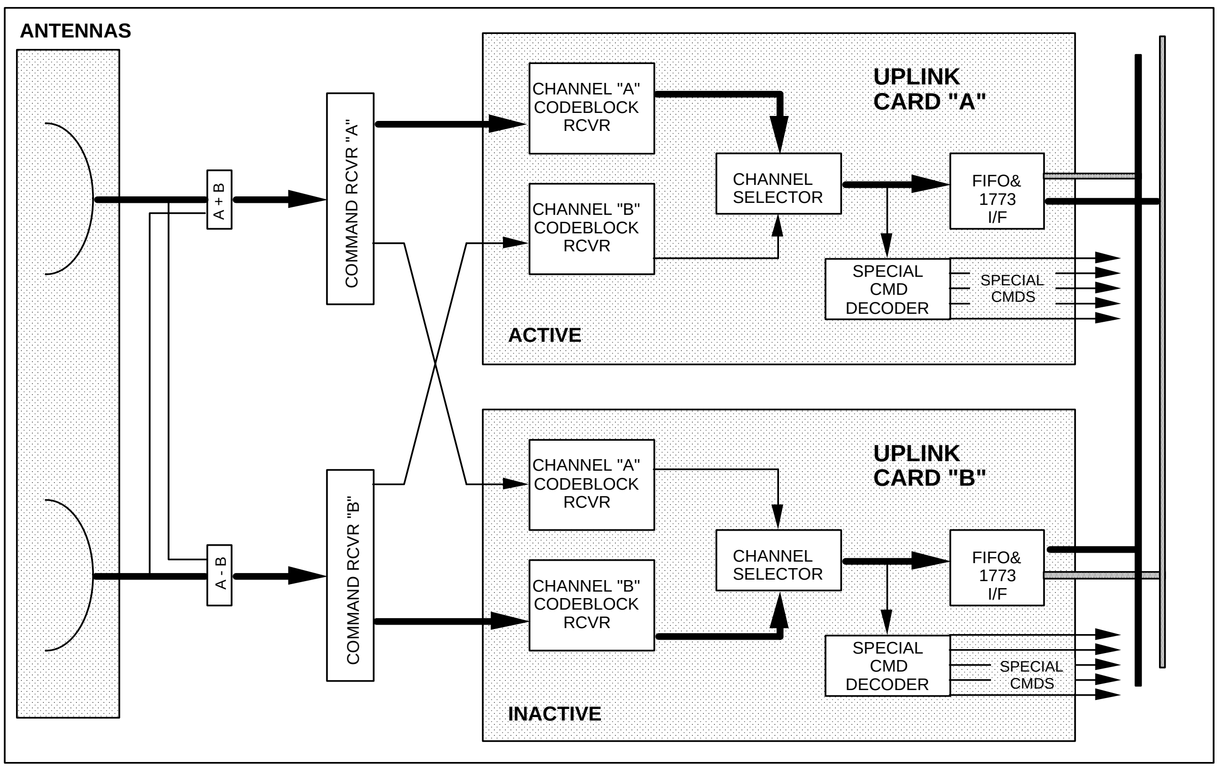


Figure 4.1-4 Uplink Cards (Normal Configuration)

The Uplink card also detects Special command data and distributes the appropriate signals. All special commands are transmitted on VC 2, and are addressed to a specific Uplink card. Special-Special commands are single code block commands which protect against a failure on the Uplink card itself, which could affect operation on the S/C bus. The Uplink cards cannot be powered OFF in flight.

4.1.1.4 Downlink Card

The Downlink card receives telemetry data across the SDS backplane, encodes the data (CRC, R/S, Pseudo-randomize, and Convolutional encoding), and outputs the serial stream to the transponder. NRZ-L telemetry is output through I- and Q-Channels, or Biphase-L telemetry output through the GSTDN/hard-line interface channel. Each channel is independently configured, and multiple data rates are supported.

The Downlink card supports clock correlation by providing a signal to the Clock card. Section 4.1.7 provides a detailed description of S/C clock maintenance.

Telemetry downlink channels are cross-strapped in the C&DH system, prior to the transponders. Each Downlink card receives telemetry data only from the SDS side it is located on and there is no direct cross-strapping of Downlink card to side A or side B processors. Telemetry can be cross-strapped from one Downlink card to the other, however, by routing data between the two Downlink cards. IPSDU-operated relays on the Downlink card route telemetry to either the in-line transponder or the alternate Downlink card for transmission via the alternate transponder.

Operationally, Downlink card B will be configured to receive telemetry from Downlink card A (prime). During normal operations, Downlink card A will be configured to send telemetry to Transponder A. During center frequency measurement supports on Transponder B, Downlink card A will be configured to route telemetry to Downlink card B for transmission to Transponder B. Downlink card B is not required to be powered ON in this configuration. This cross-strap configuration is driven by the Transponder, which can only accept one input telemetry source. Figure 4.1-5 and Figure 4.1-6 demonstrate the nominal cross-strap configuration of the Downlink cards for transmission via Transponder A (HGA) and Transponder B (Omni), respectively. Note: These diagrams assume SDS A is primary.

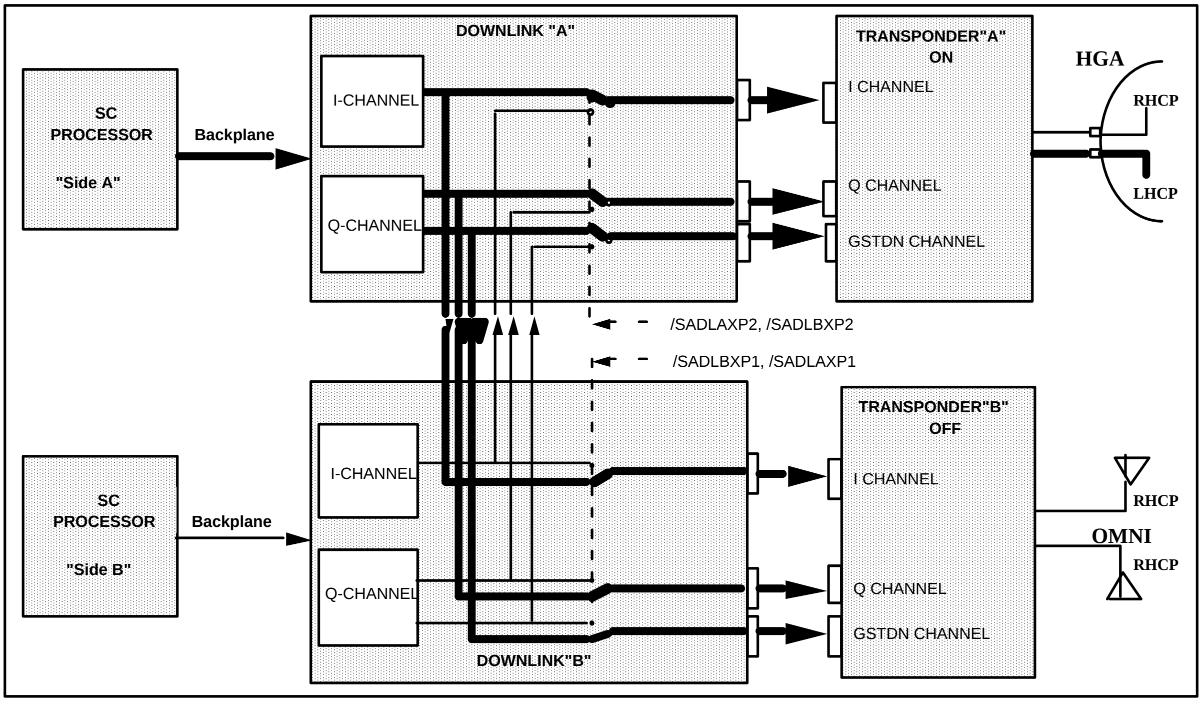


Figure 4.1-5 Downlink Card Nominal Cross Strapped Configuration (XP-A)

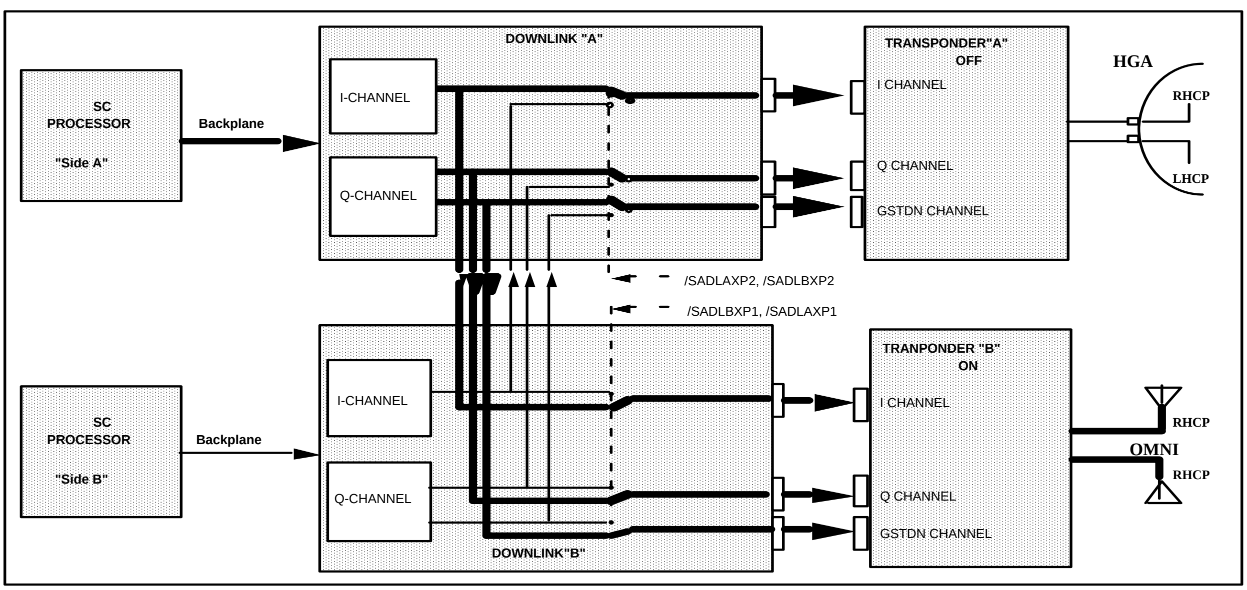


Figure 4.1-6 Downlink Card Nominal Cross Strapped Configuration (XP-B)

4.1.1.5 Clock Card and Frequency Standard

The Clock card maintains a hardware-based spacecraft time code and provides master timing reference signals for the observatory. The Clock card is driven by an externally mounted

Frequency Standard (FS), but is powered from the same power supply. The FS provides a 4.194 MHz oscillator frequency to the Clock card, which is divided down to provide the desired Spacecraft clock frequencies. Each FS provides input to one SDS Clock card only, and there is no cross-strapping.

The Clock cards require bus commands from the SPSDU to activate them. Once powered, timing signals are sent to both IPSDU A and B. The IPSDU will be configured to select the appropriate clock signal, and to provide multiple 1 Hz distribution circuits for transmission of the clock signal to observatory subsystems. Figure 4.1-7 shows available cross strapping in clock card selection.

4.1.1.6 Low Voltage Power Converters

Four LVPCs provide power conversion for each of the SDS. Each LVPC receives an input voltage of +28V and generates an output voltage of 5.05 V. In addition, each LVPC processes special (Hardware) commands to "reset" the processors. Upon receipt of the reset command the LVPC outputs a special "reset" signal, causing all components on the backplane to perform a reset operation. Table 4.1-3 provides details of the LVPCs.

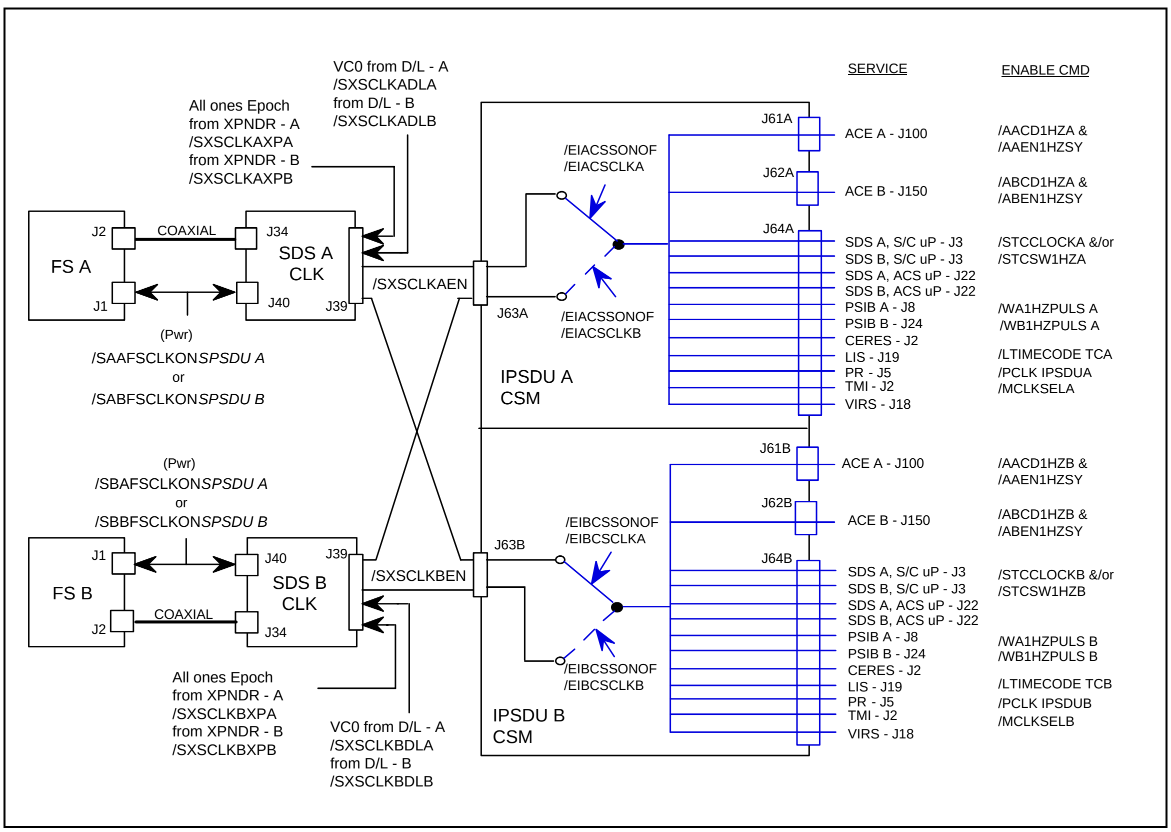


Figure 4.1-7 Clock Card Source Selection

LVPC	Components	Description
S/C Processor	S/C Processor Downlink Card Memory Cards	Card is plugged into the Backplane
ACS Processor	ACS Processor	Card is plugged into the Backplane
Uplink Card	Uplink Card	Power Converter module located on card
Clock Card	Clock Card Frequency Standard	Power Converter module located on card

Table 4.1-3 LVPC Configuration

4.1.2 Operating System

The operating system can be considered the lowest level in the FDS's layered architecture, providing various system services to the levels above. The hierarchical FDS structure is illustrated in Figure 4.1-8. The FDS operating system is supported by Ready Systems VRTX operating system, which provides a multi-tasking environment. The FDS operating system provides task and hardware initialization, mode transition services and logging, 386 and 387 exception handling, and system library services. FOT interaction with the operating system will be minimal. Higher level (application) software tasks provide an added degree of safety and convenience.

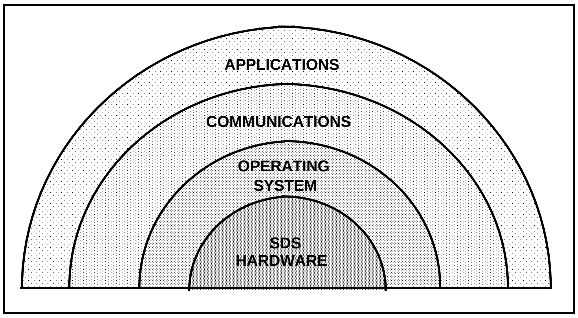


Figure 4.1-8 FDS Overview

Following a power-ON or reset during flight, the ACS and/or S/C processors are initialized to a known state. The FDS will automatically transition into one of two modes, Standby or Boot. From these modes, ground commands must be sent to affect the transition to any other valid modes or to the Hardware Diagnostics function (as described in section 4.1.2.5). The following sections highlight the characteristics and purposes of each operating system mode, describes mode transitions, and processor resets. Descriptions of the various tasks are provided in Sections 4.1.4.1 to 4.1.4.11. Table 4.1-4 provides a mode-to-task matrix for both the S/C and ACS processors.

Task & Acronym		S/C Processor			ACS Processor		
		Standby	Boot	Normal	Standby	Boot	Normal
Time Code	TC		X	X		X	X
Packet Framer	FR	X	X	X	X	X	X
Software Manager	SM	X	X	X	X	X	X
Health & Safety	HS	X	X	X	X	X	X
Software Bus	SB	X	X	X	X	X	X
Command Ingest	CI		X	X			
Telemetry Output	TO	*	X	X			
Stored Command	SC			X			X
Payload Manager	PM			X			
Telemetry & Statistics Monitor	TS			X			X
Data Storage	DS			X			
External BC-Spacecraft bus	XB-S		X	X			
External BC-Instrument bus	XB-I		X	X			
External BC- ACS bus	XB-A					X	X
External RT-Spacecraft bus	XR-S	X			X	X	X
Checksum	CS			X			X
Memory Scrub	MS			X			

ACS Task	AC						X

* See item e, in section 4.1.2.1 (Standby Mode).

Table 4.1-4 S/C and ACS Processor Mode to Task Matrix

4.1.2.1 Standby Mode

In Standby mode, power is applied to the processor with the processor configured as a Remote Terminal on the Spacecraft bus. Standby mode allows flight software re-programming and hardware diagnostics in the redundant processor(s) while the primary S/C processor is acting as Bus Controller. The FDS will automatically transition to Standby mode upon power-ON or after a cold reset. FDS services available in Standby mode are as follows:

a. Software loads and dumps through the 1773 bus as a remote terminal.
b. The transmission of status information over the 1773 bus (when requested by primary processor).
c. The transition to Boot mode through special commands.
d. Hardware diagnostics.
e. Provides 1K telemetry, but no command capability of TO task exists until transition to Boot mode.

In slightly different terms, a processor in Standby mode is running boot EEPROM code and is configured as a remote terminal on the busses with which it connects. While it is possible to configure both spacecraft processors into Standby mode at the same time, this configuration yields unpredictable results (since there is no bus controller) and should not be attempted in normal operations.

4.1.2.2 Boot Mode

In Boot mode the processor is running Boot EEPROM code with the processor configured as a Bus Controller. Basic communications and command services are provided. Boot mode allows flight software re-programming on the primary ACS and S/C processors. A key point is that the collection and transmission of housekeeping data are maintained while in Boot mode, as the processor is configured as the bus controller. The following functions are available for a processor in Boot mode:

a. Software loads and dumps.
b. Telemetry collection and downlink from selected remote terminals on the 1773 bus.
c. Transition to Normal mode.
d. Hardware Diagnostics.

It is possible to configure both S/C or both ACS processors into Boot mode at the same time. This should be avoided because cross-talk between both processors on their respective busses will impede communications.

If the primary ACS processor is in Boot mode, an ACE SafeHold condition exists, since the ACS task is not running.

4.1.2.3 Normal Mode

By definition, a processor in this mode is running Normal mode EEPROM code and is configured as a full capability Bus Controller. Upon entry into Normal mode, the operating system starts the tasks that are associated with Normal mode.

4.1.2.4 Mode Transitions

In normal operations, the primary S/C and ACS processors will be in Normal mode and the redundant processors will be powered OFF. In response to unusual conditions, the primary and redundant processors can be commanded to other modes. Transitions can be commanded from the ground or from flight software, but all are accomplished according to rules that define "legal" changes. Figure 4.1-9 shows the valid mode transitions and the effects of processor resets.

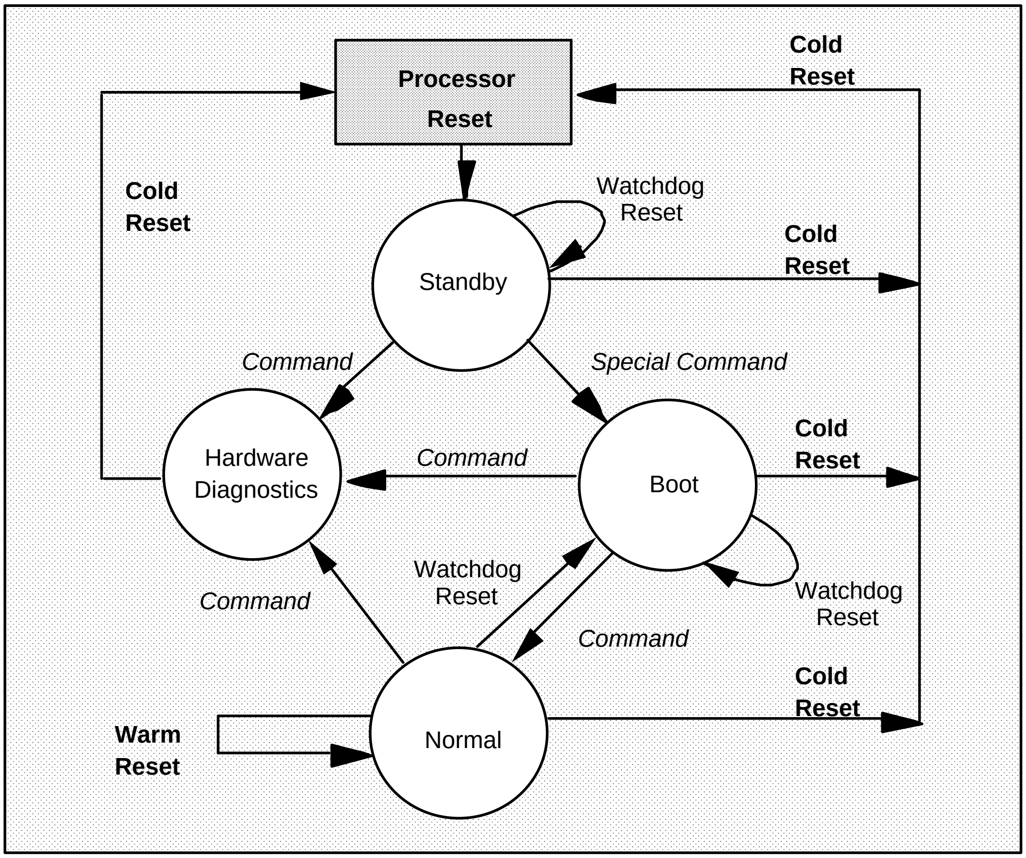


Figure 4.1-9 Valid Mode Transitions

A Special Hardware ground command is required for the transition out of Standby mode. To go from Standby to Normal, the command sets the Bus Controller/Remote Terminal bit and the FDS transitions to Boot mode. An additional command is required to transition from Boot to Normal mode.

Situations which require the FOT to command mode changes include early orbit checkout, diagnostic procedures, transition to redundant processors for operations, and recovery from anomalies which resulted in FDS-initiated transitions. Specific operations procedures identify when and how the FOT should command mode transitions.

All transitions are logged in the processor's System Mode Transition Log, which contains entries for the current and previous modes, the reason for the transition, processor context data (e.g. register data), and the transition time. Warm and cold resets are logged in the Mode Transition Log, as well. The Mode Transition Log will be dumped, as a memory dump, following any anomalous mode transition once the processor has been configured to Normal mode.

4.1.2.5 Processor Initialization

The S/C or ACS processors may be initialized under the following circumstances:

a. Power is initially applied to the processor card (a "cold start").
b. A reset Special command was received across the backplane to the processor card.
c. The watchdog timer expires.
d. An internally-generated or command requested reset is performed. This is referred to as a "cold reset" and is equivalent to a Watchdog reset.
e. An internally-generated or command requested warm reset is performed.

Processor power ON always configures the FDS to Standby mode. Upon initialization, the FDS examines the Watchdog status reset bit in the processor's Bus Control ASIC to determine if the watchdog timer expired. As a part of the power ON procedure, operations will set the Watchdog status reset bit appropriately.

4.1.2.6 Processor Resets

In Normal mode, the S/C and ACS processors and the ACS math coprocessors provide a means of "graceful" fault handling by initiating resets in response to various error conditions. Warm resets are typically attempted first and are used for less severe conditions. Warm resets may occur for variable reasons, such as 387 processor exceptions. Warm resets result in a re-entry into Present mode after code is copied from EEPROM to RAM (i.e., if processor is Normal mode when a warm reset command is issued, the processor will remain in Normal mode after the reset). While code is copied from EEPROM to RAM, data and tables in RAM are generally preserved following warm resets.

The FDS system and SCP Table of Tables also contain a flag determining, on a table by table basis, if the EEPROM version of the table is to be copied over during a warm reset. For each table update the FOT performs during the course of the mission, the FOT will update a "warm reset recovery procedure" (provided by the FSMG). This STOL procedure will recover the appropriate processor to the state prior to warm reset.

Cold resets occur after a specified number of warm resets, in response to more severe conditions, or upon powering up the processor. Cold resets reinitialize the processor and (under normal on-orbit conditions) place the processor in Standby mode, copying code and data to RAM.

Any memory or tables that had been updated only in RAM (e.g. and not in EEPROM) must be re-loaded following a cold reset. For each table and memory update the FOT performs during the course of the mission, the FOT will update a "cold reset recovery procedure" (provided by the FSMG). This STOL procedure will recover the appropriate processor to the state prior to the cold reset. Ground commands are required to transition to Normal mode.

The watchdog timer is a countdown timer which is part of the Bus Control ASIC circuitry. It triggers a processor cold reset when reaching zero. The Health and Safety task resets the timer each time it executes to prevent it from reaching zero during nominal operations. The watchdog timer is used to reset the system when the System is "hung" and prevents the Health and Safety task from resetting the Watchdog timer.

If the watchdog timer expires, the FDS will check an internally stored variable to determine which mode (Standby or Boot) the processor should transition to. If the watchdog timer did not expire, the FDS enters Standby mode. In Standby mode, the BC/RT bit is polled periodically to determine if the transition to Boot mode should be made.

A processor in Normal or Boot mode will be configured to Boot mode after a reset. A processor in Standby mode will be configured to Standby mode after a reset. All resets are logged in the System Mode Transition Log. Table 4.1-5 provides a summary of reasons to reset the spacecraft processor, as well as the boot-up mode of the processor after the reset command.

Command to Reset Processor	in STBY	in Boot	in Normal
Power OFF/ON	STBY	STBY	STBY
Special Reset Command	STBY	STBY	STBY
Watchdog Reset	STBY	Boot	Boot
Cold Restart Command	STBY	Boot	Boot

Table 4.1-5 Processor Resets

4.1.3 Communications

The communications layer provides processor independent and user-transparent data routing and distribution among spacecraft subsystems, between SDS components, and among FDS tasks. The following paragraphs discuss external and internal FDS communications.

4.1.3.1 1773 Communications

For communications with the rest of the observatory, the C&DH subsystem uses remote terminals (RTs) to interface with the central spacecraft processor via a MIL-STD-1773 data bus. The MIL-STD-1773 protocol is a command-response time division multiplexing system, which provides for the use of fiber optics as a transmission medium. In the command-response protocol, data can be transmitted only when it is requested in a master/slave arrangement. Bus Controllers (BCs) are responsible for the flow of information on the bus, and command the other terminals to transmit their data at the proper time. Remote Terminals (RTs) translate 1773 commands into parallel, 16-bit words which are processed by the target subsystem, and translates subsystem data into 1773 format for transmission over the bus.

SEUs are expected during transmission across the 1773 data busses at a rate of one every 4.66 days. Data retransmission protocols are used to minimize data losses.

Three 1773 data busses provide communications among spacecraft subsystems. These three data busses are designated as the ACS bus, the S/C bus, and the Instrument bus. In the FDS, 1773

Scheduler software is used to interface between the ACS and S/C processors and amongst the various subsystems. The 1773 Scheduler, as the name implies, conducts bus operations according to a specific I/O timing schedule, allowing for interrupts to service asynchronous processes. The 1773 Scheduler comprises separate software modules residing on specific processors performing BCRT functions. Table 4.1-6 summarizes 1773 Scheduler software modules.

Software Module	Acronym	Processor	1773 Data Bus
1773 Bus Controller	XB-S	S/C	Spacecraft 1773
1773 Bus Controller	XB-I	S/C	Instrument 1773
1773 Bus Controller	XB-A	ACS	ACS 1773
1773 Remote Terminal	XR-S	ACS	Spacecraft 1773
1773 Remote Terminal	XR-S	S/C	Spacecraft 1773

Table 4.1-6 1773 Scheduler Task Summary

In communicating over the 1773, various protocols may be used. In communication with the instruments, CCSDS packets are received over the bus from the instruments, while raw data samples from the PSDUs are read and packetization services are provided by the FDS. Once data has been received by the scheduler, it is passed to the FDS Software Bus for routing.

1773 data communications only support 16 bit data transfers across processors. However, often only 8 bits of data are required for transfer and the remaining 8 bits would be "fill" data. To maximize efficiency in inter-processor bus traffic, the capability to send multiple packets within a bus timing slot is provided through the Packet Framer (FR) task. FR packages groups of packets into special frames called "framer packets." These are transmitted (typically from the ACS processor to the S/C processor) and exploded into the original packets and processed by the Software Bus. Error recovery due to transmission errors is provided. Framer operations is transparent to ground operations.

4.1.3.2 Payload Manager

The Payload Manager (PM) task provides CCSDS segmentation support needed for instrument science and diagnostics packets. Packet segmentation is required due to 1) data packets from the CERES, PR, TMI, and VIRS instruments can be larger than the 1024 Byte maximum supported by the FDS 1773 Scheduler, and 2) ground support equipment does not support CCSDS segmentation. Segmented packets (of up to 1024 Bytes) are received from the instruments by the 1773 Scheduler and routed to the PM software for desegmentation into packets of up to 8192 Bytes. These desegmented packets are then routed for storage and playback to the ground.

The PM task also provides a fix for an error in the PR SCDP that occassionally causes the seconds portion of the PR time-tag to "get ahead" of the subseconds portion (for example, 99.79, 100.39, 101.99 instead of 99.79, 100.39, 100.99). The PM fix only corrects the error on the PR science packet (the error still exists on the PR hk packet). Upon each turn-on of the PR instrument, a FDS command to turn time tag correction on for PR must be sent.

4.1.3.3 Software Bus

The Software Bus (SB) provides communication among FDS tasks, allowing tasks to send and receive packets transparently. Routing data packets transparently to the user/receiver isolates

FDS tasks from operating system dependencies. Further, the SB provides a command and telemetry buffering mechanism.

SB operation is table driven, in which any packet may be routed to any destination, mapped by the CCSDS Application ID. Upon receiving a data packet, the SB validates the sender and destination, and passes the data along. The mapping of valid sender to destination is referred to as a stream. Multiple destinations per packet is supported. Figure 4.1-10 illustrates FDS inter-processor and inter-task communication.

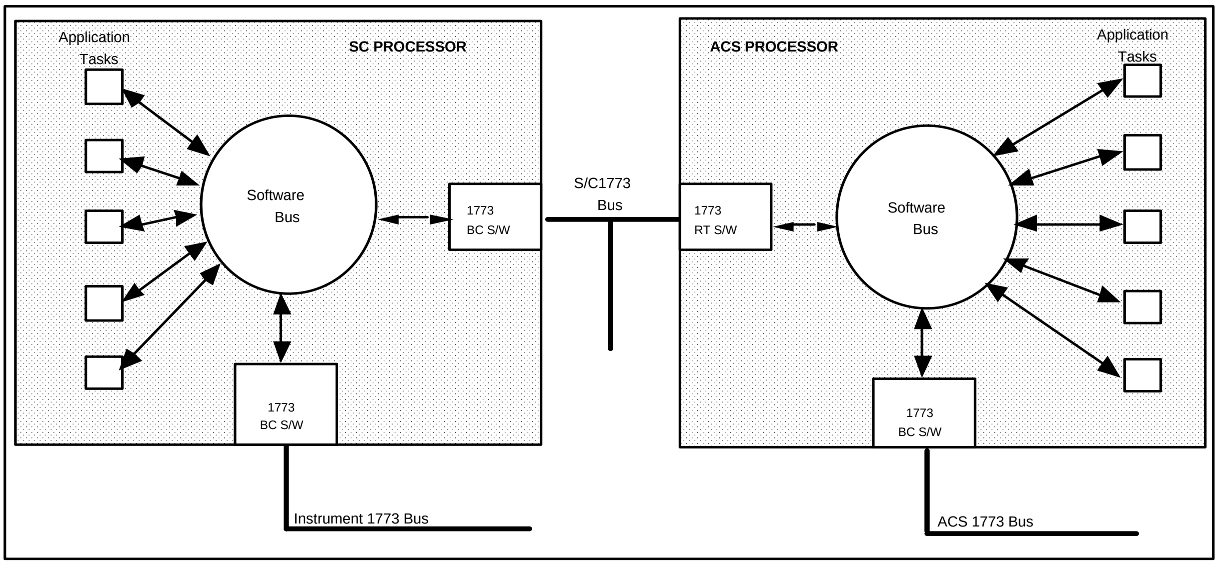


Figure 4.1-10 FDS communications

4.1.4 FDS Tasks

General C&DH functions are supported by a variety of FDS tasks. Table 4.1-7 contains the breakdown of FDS tasks to C&DH functions.

The FDS tasks Packet Framer, 1773 Scheduler, Payload Manager, and Software Bus have already been discussed, since, along with the operating system, they provide services to support the other tasks. Note that additional FDS tasks on the ACS processor are not required to support ACS software operations. The following sections provide information on the remaining FDS tasks.

Functions	Tasks	Acronym
Command	Command Ingest Stored Command	CI SC
Telemetry	Telemetry Output Data Storage Memory Scrub	TO DS MS

Communications	Packet Framer Software Bus 1773 Scheduler	FR SB XB-S XB-I XB-A XR-S
Clock Maintenance	Time Code	TC
Health and Safety	Health and Safety Checksum Telemetry/Statistics Monitor	HS CS TS
C&DH Support	Software Manager	SM
Instrument Support	Payload Manager	PM

Table 4.1-7 FDS Function-to-Task Map

4.1.4.1 Command Ingest

The FDS Command Ingest (CI) task, along with redundant sets of uplink hardware, supports receipt and processing of ground commands in accordance with Frame Operations Protocol-1 (FOP-1). Details regarding MOC real-time command generation and uplink can be found in Section 6.2. CI performs processing on the code block, transfer frame, and packet levels of command data to ensure successful transmission to the spacecraft. COP-1/FOP-1 is a protocol in which the spacecraft verifies data received correctly, received in order, and when necessary, requests retransmission from the ground system.

TRMM implementation of the CCSDS AOS recommendations assures:

a. One command packet per transfer frame. There is no plan to logically group commands and still ensure correct acceptance by the spacecraft.
b. Command packets do not span transfer frames. The largest TRMM command will be 250 bytes.

4.1.4.1.1 Code block Processing

Telecommand Codeblocks are 64-bit blocks consisting of 56 data bits, 7 checksum bits, and 1 parity bit. Code block parity validation is performed by hardware. The FDS discards any code block marked as invalid or any code block that has failed parity verification. The last 50 Codeblocks are kept in a "trash can" buffer that can be dumped for analysis. Since a transfer frame can span several Codeblocks, a bad code block can prevent complete transfer frame reconstruction. When this occurs, the partial transfer frame is discarded along with the offending code block.

4.1.4.1.2 Transfer Frame Processing

Frame level validation consists of the following transfer frame checks: version number, spacecraft ID, virtual channel (1 or 2), frame length, and sequence number. Sequence number validation failure results in command discard and further actions are possible depending on the type of failure. Valid command frames may be divided into types as shown in Table 4.1-8.

Command Type	Virtual Channel	Description
Acceptance Data Frame	VC 1	Spacecraft commands in non-bypass mode
Bypass Data Frame	VC 1	Spacecraft commands in bypass mode
Bypass Control Frame	VC 1	CCSDS Control commands
Bypass Data Frame	VC 2	Special (Hardware) commands

Table 4.1-8 Command Frame Types

All normal spacecraft command activity will use COP-1/FOP-1 protocol in the non-bypass mode to ensure correct reception by the spacecraft. CI validates the received Telecommand frame, and, if it is a valid frame, determines possible actions based on the frame sequence number. The received frame sequence number is compared against the range of valid values from the sliding window, as shown in Figure 4.1-11.

The sliding window tracks the next expected transfer frame sequence number and a range of 126 possible other sequence numbers, 63 less than expected or 63 more than expected. If a frame sequence number outside of these ranges is received, the spacecraft enters the lockout mode. In lockout mode, no further commands are processed until the spacecraft command VC has been unlocked by a CCSDS Control command. If the received frame's sequence number is in the "63 ahead" range, the assumption is that command frames were "lost" or damaged and they should be re-transmitted. If the sequence number is in the "63 less" than expected range, the frame is discarded. This second situation can happen following a telemetry outage, which could result in an incorrect next expected sequence number on the ground. The sliding window technique is a means of implementing a go-back-N communication protocol.

The FDS reports transfer frame receipt and processing status in the Command Link Control Word (CLCW). Items included in the CLCW include the next-expected-sequence number, the retransmit, lockout, and bypass mode flags.

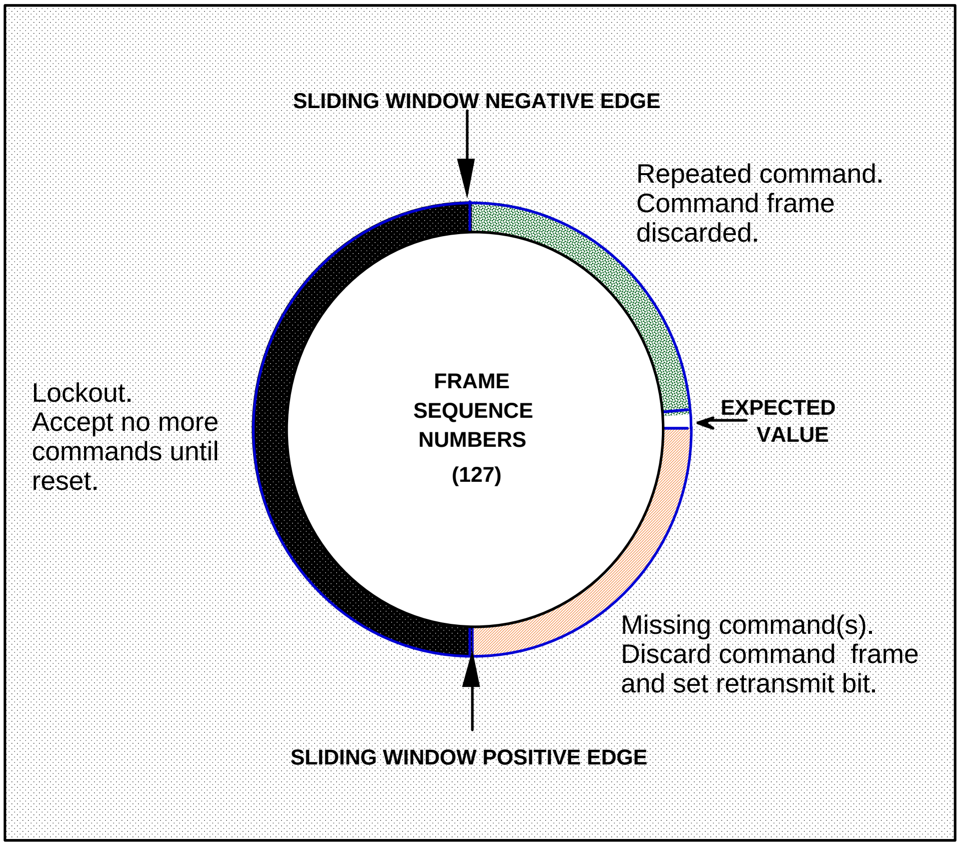


Figure 4.1-11 COP-1/FOP-1 Sliding Window

The Bypass command mode controls the settings used by CI to process commands for VC 1. Bypass is a mode that is used primarily to send CCSDS Control Commands and Special commands, however, any valid spacecraft command may be sent in the Bypass mode. TRMM's CCSDS Control Commands are "Set Next Expected Sequence Number to Zero" and "Unlock." Operationally, bypass commanding will only be used to recover from an anomalous situation, blind acquisitions, and to transition to normal command operations.

Special (hardware) commands are formatted differently by the ground system and are decoded by SDS hardware rather than software. These commands are typically used for processor selection, uplink card selection, and other hardware configurations. All Special commands will be transmitted on VC 2 in the Bypass mode. CI will generate an event message for each Special command processed, with the exception of the processor reset command on which CI is running.

4.1.4.1.3 Packet Processing

Packet level processing includes validation and exclusive-OR'ing of the packet contents. Command data is exclusive-OR'd for uplink to ensure a sufficient bit transition density to maintain transponder lock. Exclusive-OR'ing the data, again following receipt on the spacecraft yields the original command data. Packet header field formats, packet lengths and optionally packet checksums are checked, and any packets failing validation are discarded. The FDS keeps a copy of the last discarded packet in a "trash can" buffer for analysis and generates an event message describing the occurrence. Valid commands are passed to the Software Bus for distribution, with the exception of CI tasks commands which are processed immediately.

4.1.4.2 Stored Commands

The FDS provides for command execution without real-time ground intervention via the Stored Commands (SC) task. Sequences of commands can be pre-loaded along with their execute times, which can be stated in absolute UTC or relative times. Since many of TRMM's instrument and spacecraft activities will be conducted via stored commands, this subject is covered in section 4.1.6.

4.1.4.3 Telemetry Output

The Telemetry Output (TO) task collects and formats data from spacecraft subsystems and the instruments, and configures the downlink hardware in preparation for transmission. Data to be downlinked includes real-time housekeeping (engineering) data and recorded housekeeping and science data.

TO provides a filtering capability for each APID which allows defining a downlink rate different than the rate of telemetry generation. Filtering can be based on packet sequence numbers or on packet times. Filtering by packet sequence number provides for "n of x" frames to be downlinked. Filtering by time allows the collection of packets to be tied to the CCSDS time stamp, which can then allow meaningful correlation among packets related by time. A look-up table contains the filtering values for each APID. Operationally, specific TO filter tables will be employed based on the type of real-time support required, and will be selected by commands executing from the SCP for each support. In addition, TO supports multiple look-up tables, which are selectable by ground command.

Other functions of the TO task include packet formatting known as squeezing and splitting. The splitting capability will not be used on TRMM. Squeezing is a process that eliminates unused, high-order bytes in the telemetry stream. Certain devices, such as the PSDUs, provide data 8 bits at a time. Since TRMM uses 16-bit busses, every other byte from these devices is empty. TO "squeezes" the telemetry from these designated APIDs and eliminates this overhead.

The TO task provides a function unique to TRMM's Virtual Recorder 7 (VR7), called the event buffer, which is used to record flight status messages. These messages are downlinked both in real-time and from the VR, but the recorded messages are placed in the real-time stream, Virtual Channel 0 (VC0). To accomplish this, TO extracts flight status message packets from VR7 virtual channel data units (VCDUs) and places them in VC0 VCDUs. Flight status messages can be generated by the following: Spacecraft prime and standby processors, ACS prime and standby processors, ACS software, ACE A and B. To keep real-time and stored messages distinct, TO modifies the messages' APIDs. For example, messages coming from the spacecraft prime processor have APID 07 (Hex) when downlinked in real-time and APID 08 (Hex) when downlinked from the event buffer.

As event messages may be generated and downlinked during the acquisition phase of a real-time contact (when the ground has not yet acquired the spacecraft, but the spacecraft is transmitting telemetry), all accounting of event messages will be performed on event messages captured in the recorder.

In addition to managing event buffer dumps, the TO task also "handshakes" with the Software Manager tasks (one task per processor) to provide table and memory dump control. Dump packets are requested by TO, so only one table or memory dump from any one source can be in progress at a given time. TO is also responsible for suspending and resuming dumps.

Telemetry Output supports time correlation by providing information to the Time Code task and arming the downlink card to latch the sequence number of the correlation VCDU. Time code maintenance is described in more detail in Section 4.1.7.

To configure for downlink, the TO task refers to the rate combination table which contains I- and Q-Channel rates and modes. This table is expected to change little over the course of the mission. The default data rate is the 1 Kbps mode, and is used following processor cold resets.

Prioritizing the data to be downlinked is also accomplished by the TO task. Using tables, TO plays back virtual recorders in an order of priority. Additionally, transfer frames are transmitted according to the following fixed priorities:

a. Full real-time frames.
b. Re-transmit frames.
c. Playback frames.
d. Partial real-time frames.
e. Fill frames.

4.1.4.4 Data Storage

The Data Storage (DS) task is responsible for managing the operation and maintenance of bulk memory, where spacecraft and instrument housekeeping and science data are stored. Bulk memory is logically divided into segments, or virtual recorders, which are each assigned to hold data from designated APIDs. Unlike a conventional tape recorder, bulk memory can simultaneously playback and record data. To keep track of the data on each virtual recorder, the Data Storage task manages up to four partitions, called data sets, per virtual recorder. Data set 0 is the data set that is currently receiving data for storage while data set 3 holds the oldest data on the recorder.

The Data Storage task allows for two record modes, Overwrite, which overwrites the oldest data once the VR is full, and Non-Overwrite, which stops recording operations when the virtual recorder is full. The software default mode of operations is Non-Overwrite.

Operationally, TRMM housekeeping and science recorders (all Virtual Recorders) will be configured in the Non-Overwrite mode. By using the Non-Overwrite mode, data from an anomaly will be retained for later analysis in the event of a failure to acquire the spacecraft. TRMM science data may be summarized as "mapping data" where data collected over one area is not more valuable than over another area. Therefore, in order to maximize useful science data capture on the ground, the science recorders will contain the same time stamped data as the housekeeping recorder in order to use the on-board attitude data for science processing.

Continuous Playback and Non-Continuous Playback modes are also supported. In Continuous Playback mode, when playback of all recorded data has been completed, the data stream continues with real-time data. In Non-Continuous Playback mode, when a stored data playback has been completed, the downlink from that virtual recorder stops. The Non-Continuous Playback mode is the baseline operations approach for TRMM.

Filtering, which allows data to be stored at rates lower than it is acquired, is also provided. The same methods are used as in TO filtering. The memory scrub task corrects single bit errors by reading and correcting the bit and writing it back to the original location. Double bit errors are detected and reported. Check bits used in this process are generated on the first pass through memory after task initialization. Therefore, on this first pass, error detection and correction do not occur and error counts, which are not applicable, are not reported in telemetry.

Various telemetry points are available from the Data Storage task relating to bulk memory in its entirety and to each virtual recorder. Examples of telemetry relating to bulk memory include command counters and statistics. Examples of telemetry relating to the virtual recorders are data set statistics. Ground commands allow data set management, playback start/stop and pause/resume, and memory management. A complete description of recorder management and operations is in Section 4.1.8.

4.1.4.5 Software Manager

One of the FDS tasks that the FOT will be most involved with is the Software Manager (SM). This task provides the ability to manage the memories of the 386 processors on the spacecraft. The SM provides services to promote reliable, safe access to tables and memory as well as a memory diagnostic capability through its dwell function. The level of safety for table operations is accomplished by a working buffer for use during loading and dumping and load length verification. In addition, the SM can "handshake" with other tasks to ensure that tables used by these tasks are updated only at appropriate times during their execute cycles. This capability applies to tables defined in the SM's Table of Tables as non-jam loadable. The Table of Tables is essentially a catalog of tables defined for a particular processor. It includes table locations and characteristics, such as whether a table is jam-loadable and whether it is to be checksummed. The memory dwell capability allows the contents of up to 64 words (1 word = 16 bits) to be sampled up to eight times per second and downlinked in telemetry at no more frequently than one packet per second.

The FOT will make extensive use of Software Manager services since spacecraft loads will be transmitted on a daily basis. Details regarding table and memory loading and dumping operations are in Section 4.1.5.

4.1.4.6 Time Code

The Time Code (TC) task supports the mission requirements to maintain timing accuracy to within one millisecond of Universal Time Coordinated (UTC). The Time Code task distributes time to spacecraft subsystems and the instruments (via the CSM within the IPSDU), and supports time correlation and maintenance. Time correlation is based upon coordination with other tasks, such as Telemetry Output, and epoch information from the Downlink cards. Operations for time correlation and maintenance are described in Section 4.1.7.

4.1.4.7 Hardware Diagnostics

The Hardware Diagnostics software is used to assess the functionality of the spacecraft and ACS processor cards, including their 386 processors, 387 math coprocessors, and memory areas. Hardware functionality tests are performed according to table-driven test selections and parameters. The results of the tests are written to a pre-defined area of memory. This memory can be downlinked as a table or simply as memory.

Each processor contains the Hardware Diagnostics software capable of detecting errors in the processor, its memory, and in the Bus Application Specific Integrated Circuit (ASIC). Various parameters to control which diagnostic functions are performed and specific options for those functions can be loaded using memory load methods described in Section 4.1.6. Execution of the Hardware Diagnostics is accomplished by ground command to a processor in Normal, Boot, or Standby mode. Table 4.1-9 summarizes hardware diagnostic tests.

If diagnostics control information is not loaded, a full set of tests is performed and default values for parameters are used. The longest test takes up to 6.5 minutes per Bulk memory card. Results from the diagnostics tests are stored in memory as ASCII strings which may be dumped. Diagnostics will be accomplished at the request of flight software personnel, and will typically be used for anomaly investigations.

Component	Test
Memory • EEPROM • RAM • BCRT Shared RAM • Bulk	Incremental Complementary incremental Walking ones Walking zeroes User-defined pattern
Bus ASIC	EDAC functionality Subsecond timer Register use

82380 DMA Controller	DMA Transfer Programmable interval timer Register use
80387 Coprocessor	Functionality
Miscellaneous utilities	Set memory range to pattern Memory range report Read/write port Report interrupt counts since diagnostic start

Table 4.1-9 Hardware Diagnostics

4.1.4.8 Health and Safety

This task is one of three FDS tasks designed to provide detection and correction for specific conditions on the 386 and 387 processors. Along with the Checksum and TSM tasks, Health and Safety (HS) monitors various functions for failure or anomalistic behavior and takes corrective actions. This task is the highest priority FDS task, meaning that it cannot be pre-empted by any other task. Functions that are monitored are those whose failure would endanger spacecraft safety and for which corrective actions have been defined. Specific functions performed by this task are:

a. Checking for CPU hogging by other tasks.
b. Monitoring event messages for warm/cold reset conditions. Commanding resets as necessary. Performing warm resets if critical subsystems fail to respond to a request for housekeeping packets.

In addition to the above, the Health and Safety task collects and packages housekeeping telemetry from the spacecraft and ACS processors. This telemetry consists mainly of status information, including various task execution counters, reset counters, and current/previous mode information. HS telemetry is provided for all processor modes. Both the S/C and ACS processors have a set of mode-independent telemetry that Health and Safety processes in all modes. Additional parameters are added for normal mode that reflect status for tasks that run in normal mode only.

One key function of this task is resetting a processor's Watchdog timer. The Watchdog timer is a countdown timer that is part of the bus ASIC circuitry. It triggers a processor cold reset when reaching zero. The Health and Safety task resets the timer (to 8 seconds) each time it executes to prevent it from reaching zero during nominal operations.

4.1.4.9 Checksum

The second task relating to health and safety, Checksum (CS), is designed to detect any changes in static memory areas of the 386 and 387 processors. When errors in these code and data areas are detected, a flight status message is generated. The Checksum task uses a 16-bit exclusive-or (XOR) to compute check bits for each contiguous segment of memory that is to be verified. The contiguous segment size, or granularity, is 4096 bytes. The segment size ensures that all memory is checksummed once every orbit (90 min.).

At start-up, this task goes through the system and stored command Table of Tables, calculating checksums for designated areas of EEPROM and RAM. The two basic areas that are checked are code/data segments and tables. Tables that are checksummed are those considered as "static." If a checksum error is detected in RAM a flight status message is issued. The default response for a RAM CS error is to perform a warm reset. If an error is detected in EEPROM, a flight status message is also issued, but the default response is not to perform a warm reset. Following a warm reset, the Checksum task is disabled. Upon a cold reset, the task is enabled. If a Cold reset occurs, the FOT will dump the mode transition log and check the event message for the checksum values.

4.1.4.10 Telemetry/Statistics Monitor

The TSM segment of the FDS's health and safety implementation extracts specific telemetry values from specified telemetry packets and processes them according to table-defined control parameters. Only packets which are designated to be routed to TS are available for monitoring. Up to 64 monitor points can be defined on the S/C processor, and 32 on the ACS processor. For each of these monitor points, which can be a combination of telemetry points, up to four limit values can be defined. Each limit value, called a threshold, can be associated with an event message and corrective action. Typically, RTSs are executed in response to threshold crossings. The TSM task employs mechanisms to prevent event message flooding and to prevent undesirable repetition of the same corrective actions. The monitor points, thresholds, corrective actions, and other control parameters are contained in the limits control table.

A statistics table, as the name implies, maintains statistics on each monitor point. Key values include number of threshold crossings, minimum/maximum values and associated times, number of event messages generated, time of last monitor reset, and failure counts.

In addition to modifying the limits control table, the TSM task supports ground control via the following commands: enable/disable single monitor, enable/disable all monitors, reset single or all monitors, and change mode. The task can operate in a watch mode, which entails telemetry monitoring and statistics generation, but no associated actions are taken. This mode is for Integration and Test and will not be used during normal operations.

4.1.4.10.1 SafeHold

The majority of SafeHold activities are initiated autonomously by the ACS (i.e., ACE control, maneuvering spacecraft to SafeHold attitude, indexing Solar Arrays, etc...). The spacecraft processor monitors the ACE telemetry for an indication of SafeHold. Upon detection of an ACE SafeHold indication, the FDS will initiate a spacecraft RTS. This RTS will issue commands to configure the spacecraft for the 1 Kbps rate (configuration to 1 Kbps includes selecting the appropriate rate index table and setting the appropriate encoding schemes), turn OFF transmitter A (in case SafeHold occurred during real-time), turn ON transmitter B (in the TDRS mode), and

Disables the Spacecraft ATS (to prevent future stored commands from executing). SafeHold recovery will begin upon ground detection of the cause of SafeHold.

4.1.4.10.2 Low Power

Spacecraft Low Power detection is performed by both the PSIB and the Spacecraft processor. The S/C processor will provide initial detection via a telemetry monitor. An on-board TSM has been implemented to provide a warning upon detection of potential Low Power conditions. The TSM monitors the Essential Bus voltage and Battery State of Charge. In addition, the PSIB monitors the Essential Bus voltage and Battery cell voltages in order to detect a Low Power condition.

If the PSIB detects a Low Power condition, an indicator is set in telemetry. The S/C processor will detect the presence of the "Low Power" indicator and will activate a spacecraft RTS. The RTS issues a "Low Power/SafeHold" command to the IPSDU, and the IPSDU then issues the "Low Power/SafeHold" pulse to the Instruments and PSIB. Ninety seconds after receipt of "Low Power/SafeHold" discrete command, the PSIB will remove power from the Non-Essential bus. In addition to issuing the command to the IPSDU, the RTS will individually command OPEN the power relays for each Non-Essential bus component. This provides a backup to the PSIB, and also provides for a smoother recovery from Low Power mode. Note: detection of a "Low Power" condition will not change the configuration of the spacecraft RF/Communications subsystem.

4.1.4.11 Memory Scrub

Memory Scrub (MS) maintains the integrity of data stored in the solid state recorders. MS performs a read and write of each memory location. If any SEUs have occurred, the hardware EDAC will correct the error during the read, and the write will generate a new checksum to correct the error. Single bit flips are corrected, and double bit flips are detected. There is a notification of where in bulk memory multiple bit upsets are detected, however there is no notification of where in bulk memory SEUs are detected and corrected.

Memory scrub runs once every two seconds, and scrubs a fixed amount of memory each cycle. The amount of memory scrubbed is sized such that all of bulk memory is scrubbed once per orbit. All of memory is scrubbed, regardless of whether it currently contains valid data or not.

Operationally, the FOT will monitor and trend the number of occurrences of SEUs as related to orbit position.

4.1.5 Table And Memory Operations

The Flight Data System (FDS) Software Manager task provides access to memory in the S/C and ACS processors in a manner that promotes safe and reliable operations. A combination of FDS functions, uplink protocols, and ground system capabilities is designed to uplink commands and data to processor memory without corruption. This section will describe the interface between the ground and spacecraft memory and the chronology of load and dump operations. Information relating to this topic includes Section 6.2, Command Operations, and Section 7.2, Load Generation. ACE loading and dumping is covered in Section 4.2.

Much of the memory on the S/C and ACS processors can be viewed in two ways, as tables and simply as memory. A table, in this context, is a logically-related and homogeneously structured collection of data residing in memory. Any area of memory defined as a table can also be treated as memory, which has no implicit structure or meaning. Table operations provide extra levels of safety against errors by validating uplinked loads and by coordinating with other flight software tasks before altering certain tables. Memory areas that are defined as tables are accessed as tables for all normal operations. Operationally, the tables routinely loaded and/or dumped will be Stored Command Processor tables, including ATSs and RTSs.

Table load and dump operations are accomplished through handshaking with currently running FDS tasks. There is minimal impact to FDS or ACS operations in loading and dumping, and there is no requirement to transition out of normal operations to perform these operations.

The SM task runs in all processor modes, but table services are available only in Normal mode. For Standby and Boot modes, only memory operations are supported. The two major operations involving tables and memory are loading and dumping, and within each of these, there are various possibilities and options. Table and memory loads, dumps, and patches can all be used to update the contents of the Ground Reference Image (GRI). The GRI, which is covered in Section 6.4, is a MOC software tool that is used to track the contents of the ACS and S/C memories. For memory operations and system table updates, the FOT coordinates with the Flight Software Maintenance Group (FSMG). Typically, operations of this type involve flight software maintenance and/or configuration changes, which are controlled by the flight software configuration control board.

4.1.5.1 Table Operations

Table operations are conducted from a staging area in the FDS called the Software Manager's working buffer. Full and partial table loads and dumps are accomplished via this buffer, which is sized to accommodate the largest table (the ATS buffers). Table operations are initiated with one of the following commands and associated parameters:

a. Select stored command table. Parameters include Table Identifier, From_image, Commit operation.
b. Select system table. Parameters include Table Id, From_image, Commit operation.

Upon receipt of one of these commands, the SM either copies the desired table into the working buffer or fills the working buffer with zeroes. The action taken depends on the value of the From_image parameter. The possible values are:

a. Initial, which corresponds to EEPROM.
b. Active, which corresponds to RAM.

c. Null, which zero fills the buffer.

If the commit operation parameter is set to "append," the buffer is zero-filled regardless of the From_image value. The append operation may only be applied to ATS buffers. Since ATSs reside in RAM, the From_image value will always be set to "null" for uplinked loads.

The value of the commit operation determines what will occur with the information in the working buffer following a successful operation. Possible values are:

a. Replace initial, which updates EEPROM.
b. Replace active, which updates RAM.
c. Dump only, which disallows changes to RAM or EEPROM.
d. Append active, which appends new information onto a table (ATS tables only).

Table patch loads update specific parameters within tables, without requiring that the entire table be updated. Table patch loads are described in section 6.

4.1.5.1.1 Table Dumps

Dumping tables is accomplished primarily for two reasons:

a. Retrieving status data from the spacecraft that is not downlinked in housekeeping telemetry.
b. Verifying table contents, possibly following load operations.

Dumps are initiated by a select table command with the appropriate parameters to indicate the table and source (EEPROM or RAM). A dump table command follows the select command, indicating where the dump should start, how many words to dump, and how many copies should be dumped. For full tables, the dump should start at the beginning of the buffer (offset = zero) and continue to the end of the table. The number of copies will be defaulted to a value of one by the ground. By specifying a range of values using the offset and number of words, partial tables can be dumped. The FOT can stop table dumps by issuing the abort_dump command. Table dumps in progress when a pass is terminated (i.e., when Telemetry Output stops downlinking data) will be autonomously aborted.

Table dumps can be compared against loads, the GRI, and other dumps. Real-time messages and displays contain miscompare information, including number and location of miscompares. Dump reports include additional information, such as locations and values in the dump and comparison images. Reports containing interpreted table contents are also provided by MOC software. PDB table definitions provide the formats and descriptors for each tables' entries. Dumps can be transferred to the OST for report generation and for subsequent transfer to the FSSB, which is covered in Section 6.5.

4.1.5.1.2 Table Reset

A reset command resets all table operations and values pertaining to the working buffer. This command is nominally sent after a dump and can be sent to terminate a table operation without affecting normal operations. Following a successful load and commit sequence, a reset is not required.

4.1.5.2 Memory Operations

This section describes the memory operations that apply to the S/C and ACS processors. These operations are available as a back-up access method for those areas defined as tables and as a means for accessing memory areas that are not table-defined. Some operations are similar for memory and tables (loading, dumping, and patching), but the validation that is available for table operations does not apply. Memory loading and dumping do not make use of the SM buffer, so all operations occur directly to and from memory locations. Unique memory management features are the dwell and copy capabilities, and hardware I/O diagnostics.

Memory load operations will be determined by the FSMG in response to specific conditions. For the large-scale uplink of new code, the typical operation will be to load the new code to an unused portion of memory (if sufficient unused memory exists), and then to uplink a "branch instruction" to RAM causing software to execute the new code. To support the loading of the branch instruction a non-interruptable RAM memory load command is supported. Upon receipt of this command, FDS interrupts are disabled immediately prior to command execution, and enabled immediately after the RAM load has been successful. No checks are made by the SM if the memory address specified in the command is in RAM, but will fail if this instruction is made on EEPROM.

4.1.5.2.1 Memory Dumps

Memory dumps require one command, which specifies a start address, number of words, and number of copies. Memory type is not required, since any memory area is valid for dumping. The "number of copies" parameter will be set to one by MOC software. Dumps that are in progress at contact termination are handled the same way as table dumps, with abort commands either from the ground or from the Telemetry Output task. Memory dumps can be compared against loads, the GRI, or other dumps. The MOC will report the number of compares and miscompares in displays and reports. Additional information available in reports includes locations and values for the dump and the image with which it is compared. For areas that have table definitions, MOC software can also provide interpreted memory reports. Memory dumps can also be transferred to the OST for analysis and from there can be transferred to the FSSB.

4.1.5.2.2 Memory Dwell

A capability available from SM's memory management services allows the monitoring of selected memory locations. This has the affect of turning memory address locations into telemetry mnemonics. At specified intervals, the contents of the requested memory locations are read and the contents are placed in a dwell packet. Up to 64 words can be sampled up to 8 times per second. The 64 locations can be anywhere in memory and need not be contiguous. The delay times can be specified for each location, but are relative to the previous time. In other words, if a dwell is commanded for 4 words, and each has a 1 second delay, then each word will be sampled every 4 seconds. Dwell packets, containing the values, are issued for downlink at various rates, depending on set-up parameters. Dwell packets can be downlinked no faster than one packet per second. Memory dwells may be requested by the FOT, engineering, and flight software personnel, usually to investigate unexpected spacecraft performance or to monitor parameters during special operations.

4.1.5.2.3 Memory Copy

The memory copy capability can be used to copy contiguous memory areas from RAM or EEPROM on a processor to another area on the same processor. One copy command can move up to 2048 words from RAM to RAM, from RAM to EEPROM, or from EEPROM to RAM. Extra memory loading procedures can be avoided by using the copy feature in situations where the same data needs to be updated in RAM and EEPROM. This feature can also be used when the ACS or S/C processor is in Boot or Standby mode since table operations are unavailable in these modes. If the memory copy function is used, the FOT is responsible for ensuring that the GRI is updated to reflect the changes. This is accomplished by dumping the target memory area and overlaying the dump contents onto the appropriate GRI locations. Similarly, if the copy destination includes table-defined areas, the FOT is responsible for ensuring that the GRI checksum table is updated. This is accomplished by dumping the spacecraft checksum table and overlaying the GRI with the dump. As an added assurance prior to the overlay, the FOT will typically compare the dumped checksum table with the GRI version to ensure that only the updated area miscompares. Note that code in EEPROM is compressed and therefore memory copies between EEPROM and RAM are not expected.

4.1.6 Stored Command Processing

The Stored Command (SC) task provides the capabilities to store and execute commands for instrument and spacecraft operations. Both absolute and relative time tagged commands are supported. Both the ACS and S/C processors contain the Stored Command task software, both of which are used for normal operations. Control and coordination over the execution and timing of ATSs and RTSs are maintained by the Stored Command task. Many of TRMM's operations execute from stored commands, including communications configuration sequences, science instrument control, and contingency sequences.

Any valid command except Special commands can be included in stored command loads, but the manner in which stored sequences are used depends on the purpose and execution criteria of the commands in question. The decision to store commands as opposed to issuing them in real-time is often based on real-time availability. Other criteria include the degree of precision for execution timing and the need for critical Safing sequences. The same considerations can also be used to determine whether command sequences should be RTSs or ATSs. The following guidelines help illustrate when RTSs are used:

a. A fixed sequence needs to be executed at some repetitive event (Eclipse entry).

b. Commands need to be triggered by FDS tasks (Low Power mode configuration).
c. Contingency sequences may be required at unpredictable times (Safing sequences).

RTSs may be initiated by ground command, ATS commands, FDS tasks, and other RTSs. The total number of stored commands, ATS and RTS combined, that may execute is eight per second per processor. Any commands in excess of eight will execute at the next opportunity, the time of which depends on other commands to be executed and a pre-defined priority scheme. The stored command priority scheme places ATS commands first, followed by RTSs ordered by number--the command(s) from the lowest numbered RTS executes before other RTS commands.

It is possible for commands to execute "out of order" since a delayed command with an earlier time may have to wait for higher priority commands. MOC software verifies that the total number of ATS commands and known RTS commands is eight or less. Operationally, the FOT plans to have a 1 second delay between commands in any command sequence, as applicable. Known RTS commands are limited to the RTSs that are started from the ATS or chained RTSs that were initiated by the ATS, and do not include those started in real-time or by TSM. Commands could still be delayed since FDS tasks and real-time operations personnel can also execute RTSs. Restrictions that apply to either RTSs or ATSs will be covered in their respective sections.

ATS loads are built by MOC software and are based on commanding requested in Daily Activity Plans (DAPs). RTSs are submitted as separate files, but the load generation request is included in the appropriate DAP. The planning and load generation processes are covered in detail in Section 7. The term "stored commands" in this section refers to PDB-defined commands stored on board for execution at a future time.

4.1.6.1 Load/Verify Operations

ATSs and RTSs are managed as tables for loading, verifying, and patching processes as described in Section 4.1.5. As with other tables, stored command loads consist of a series of load_table commands, each containing absolute or relative time-tagged commands in its data segment. The data portion of each command contains 200 bytes of command data, except the last command in the load. The last command typically contains the remaining data and fill data to total 200 bytes. RTS and ATS loads are verified by word count comparison during the load process and each command is validated at execution time. This validation is based on the checksum in the command packet's secondary header. Unique features of RTS and ATS operations and the load structure are described in the following sections.

4.1.6.2 ATS Load Processing

ATS commands require a command number and time tag, in addition to the executed command. The command numbers in the ATS commands are used only to identify commands, and are not for command ordering. These numbers are used in controlling ATS processing and in patching loads once they are in the ATS buffers. The time tag represents time in UT format. The format of ATS commands and ATS command loads in memory are shown in Figure 4.1-12.

The SM accepts the load_table commands and places the data contents (the ATS commands) in the working buffer. Upon successful word count compare, the SM notifies the Stored Command task that the new load is available, and after an acknowledging handshake, the load is copied to the designated ATS buffer.

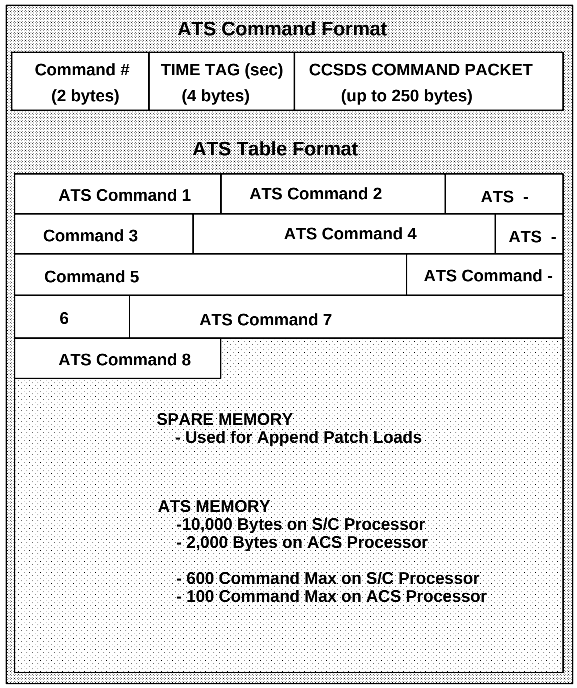


Figure 4.1-12 ATS Command and Load Format

4.1.6.2.1 ATS Load Modifications

There are several ways to modify the contents and processing of an executing ATS load. The choice of method used in a given situation depends on the amount of preparation time available and the number of changes required. In general, off-line methods, such as regeneration of a load are chosen when time allows and when many changes are required. Changes of an immediate nature can be accomplished with patch loads, which use the stored command "append" function. The following methods are available for altering stored command processing:

a. Load other buffer and command switch_buffer in real-time.
b. Build patch load and append to active buffer.
c. Send "jump" command to skip undesired commands.
d. Send "stop_ATS" command to prevent execution (only in extreme conditions, e.g. to allow troubleshooting).

TRMM operations procedures provide guidelines on when each of the above methods should be used. A more detailed description of MOC load processing is provided in Section 7.2.3.

4.1.6.2.2 ATS Processing

Upon activation of an ATS buffer, the Stored Command task builds a number of data tables that are used to control ATS execution. All sequencing is done at this point, so loads need not be in chronological or numerical (by command number) order. The structures built by the Stored Command task to manage the execution of the ATSs are:

a. Processor control block: lists current buffer and command number, process state, command counter.
b. ATS status table: lists status for each command (empty, loaded, executed, skipped, failed).
c. Time index table: contains chronologically ordered list of command numbers.
d. Command index table: contains pointers to commands associated with the command numbers in time index table (therefore in execution order).

At start-up, which results from a start_ATS or switch_buffer command, any commands with expired time-tags are skipped. An append operation essentially re-starts the buffer such that the control structures are rebuilt. The ATS processor (under the auspices of the Stored Command task) executes one command at a time by fetching the command, verifying the secondary header checksum, and if successful, sending the command to the Software Bus. The appropriate control structures are updated to reflect this, and processing for the next command can begin.

Patches to an ATS load, which use the append option of select_storedcommand_table, can be used to replace, remove, and add commands that are in an ATS buffer. Replacing commands is accomplished by assigning the new command(s) the same number(s) as those to be replaced. Command removal is done similarly, by assigning a No_Operation command to the same command number as the command to be removed. For adding commands, numbering can start where the original load's command numbers end. Since command numbers are not used for ordering, these added commands can be inserted anywhere within the load. MOC software keeps track of command numbers and assigns them for the appropriate action, replacement, removal, or addition of commands. However, it is the FOT's responsibility to ensure that the switch_buffer command is managed appropriately during append operations.

4.1.6.3 RTS Operations

The S/C processor provides storage and management for 128 RTSs while the ACS processor supports 24. RTSs are numbered, 300-byte tables, and are prioritized according to number, with RTS 1 having the highest priority. RTSs used for Safing the spacecraft in contingency situations are assigned the high priority low-numbered tables while housekeeping and instrument RTS are located in high-numbered tables. Command sequences that are longer than 300 bytes can be split, or "chained" into as many RTS tables as required. The Start_RTS command is appended to the end of each RTS in the chain except the last one.

A time tag representing a delay in seconds prior to executing the CCSDS command is included in the RTS. Figure 4.1-13 details RTS command and table load format.

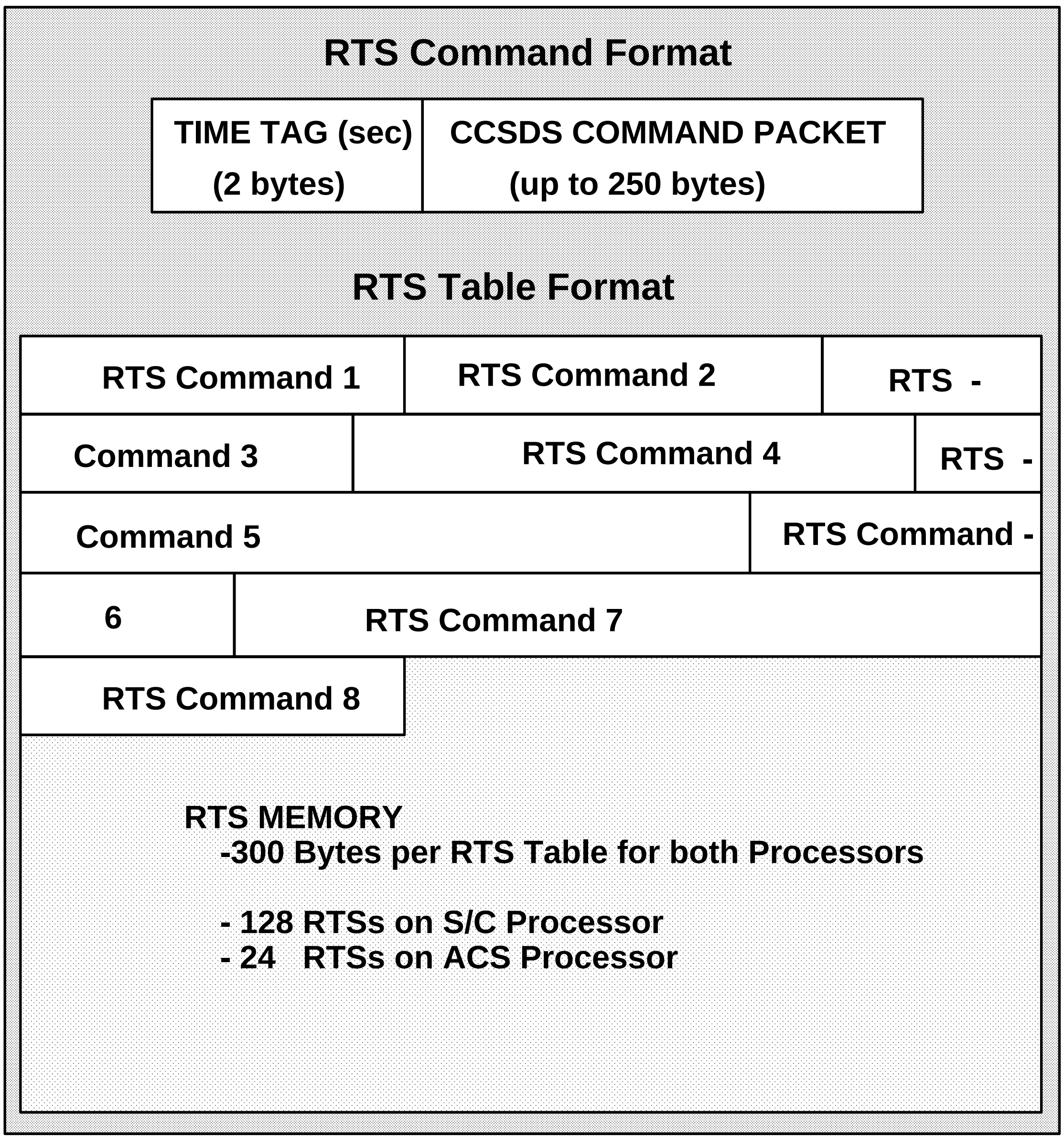


Figure 4.1-13 RTS Command and Table Format

4.1.6.3.1 RTS Load Modifications

The methods for altering the course of RTS processing include disabling the RTS, re-loading the RTS, or inhibiting the RTS call (e.g. modifying TSM or ATS load). In contingency situations, the most expedient method is to disable the RTS.

4.1.6.3.2 RTS Processing

The RTS processor, a subset of the Stored Command task, manages the execution of RTSs. All of the RTSs on a processor (S/C or ACS) can be active at a given time, but like the ATS processor, only one command executes at a time. By keeping the number of commands at eight or less per second, the Stored Command task can ensure that command timing resolution requirements are met. The RTS processor builds and maintains a control structure called the RTS Information Table, which contains the following:

a. Status of each RTS, enabled/disabled and active/idle.
b. Next command execute time.
c. Counters: commands, errors, activations.
d. Next command pointer.

A number of statistics relating to RTS processing are maintained in a table that can be dumped for housekeeping and analysis purposes. For each RTS, entries will include the number of activation failures and successes, and the number of command failures. The real-time telemetry stream will contain information from the RTS information table, such as enabled/disabled and active/inactive status, and the number of the last RTS that failed.

4.1.7 Spacecraft Clock Operations

The FDS includes redundant spacecraft clock and timing systems. Each timing system is comprised of an oven controlled master oscillator also known as the Frequency Standard (FS), a Clock Card, and a series of timing circuits. The following paragraphs describe the TRMM timing system and its operation. Figure 4.1-14 provides a block diagram of the TRMM timing system.

4.1.7.1 Frequency Standard (FS) Operation

The master oscillator will, for all remaining reference, be termed the Frequency Standard (FS). The TRMM timing system employs two FSs. Nominally, only the FS associated with the prime FDS will be powered. The second FS provides redundancy and will only be powered in the event of a failure with the prime unit. The FS outputs a frequency reference of 4.194 MHz to the timing chain. The 4.194 MHz frequency reference output by the FS is divided by the timing chain into four clock signals: 1.048 MHz, 32 Hz, 1.0 Hz, and 64 seconds. Only the 1Hz signal is distributed to the TRMM spacecraft subsystems and instruments via the IPSDU.

The FS maintains frequency stability by use of temperature-controlled inner and outer ovens employing thermostatically controlled heaters. After approximately 45 days of continuous operation at any constant temperature between $+5^{\circ}$ and 50° C for the A-Side and -10° and 50° C for the B-Side, the long-term frequency stability of the FS is expected to be better than 5×10^{-8} Hz. However, to compensate for inherent drift, the FS can be offset by ground command to ±1.0 Hz about the 4.096 MHz center frequency.

As part of the launch configuration sequence, Project Integration and Test (I&T) personnel will set the FS offset to zero drift rate, which is equivalent to the nominal 4.096 MHz center frequency. Once on-orbit, the FS will be monitored and maintained by the FOT. Because the FS frequency is not provided in telemetry, TRMM mission operations must use the spacecraft clock as an indication of the FS output. By monitoring the spacecraft clock and trending the computed delta time with respect to ground UTC over a period of time, conclusions of FS performance can be drawn. For example, if the spacecraft clock continually runs faster than the ground UTC, it can be assumed that the FS is running at a frequency higher than the nominal 4.096 MHz value and should be adjusted to a lower value to effectively slow down the spacecraft clock. Likewise, if the spacecraft clock continually runs slower than the ground UTC, it can be assumed that the FS is running at a frequency slower than the nominal 4.096 MHz value and should be adjusted to a higher value to effectively speed up the spacecraft clock.

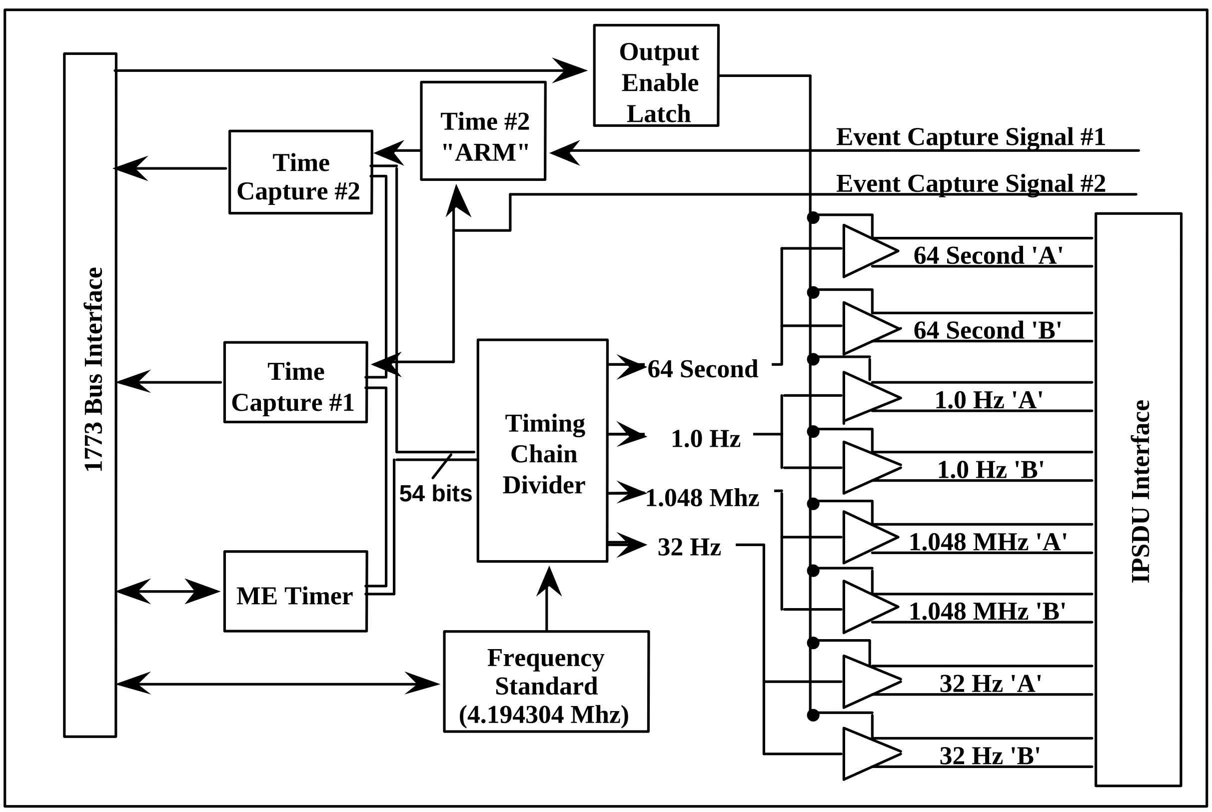


Figure 4.1-14 Timing System Block Diagram

4.1.7.2 Spacecraft Clock Description

The FDS maintains the S/C clock in two parts, a Spacecraft Time and a Universal Time Correlation Factor (UTCF). To arrive at actual UTC, the Spacecraft Time and the UTCF are summed together. This summation will provide the number of seconds since the mission epoch (equivalent to January 1, 1993). Both the Spacecraft Time and UTCF are formatted as 64-bit fields, with the most significant 32-bits representing seconds, and the least significant 32-bits representing subseconds, as shown in Figure 4.1-15. It is important to note, however, that the first 20 bits of the subseconds field (0.953674 µsec) is the smallest time unit in the hardware clock. All other clocks may use the full 32 bits. The FDS Time Code (TC) software updates the Spacecraft Time 'Seconds' field by reading the hardware clock value in synchronization with the 1 Hz timing pulse. The 'Subseconds' field is taken from the Bus Control ASIC by the TC task and is also relative to the 1 Hz pulse.

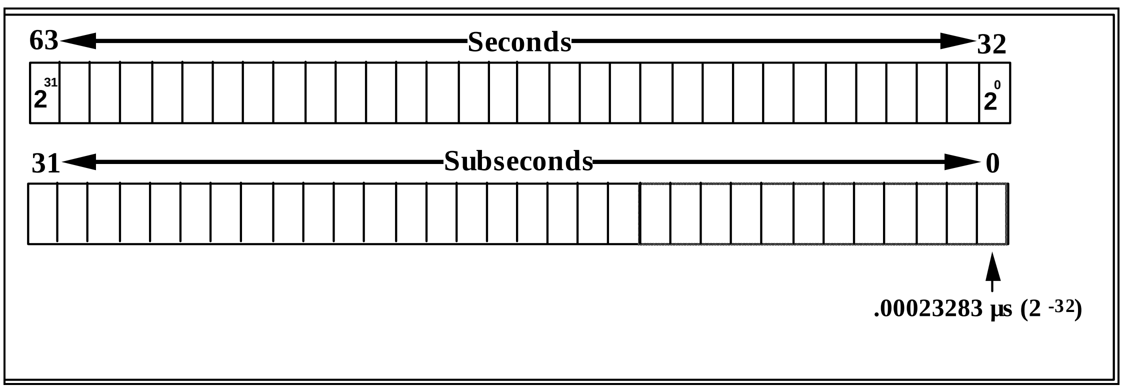


Figure 4.1-15 S/C Time Code and UTCF Format

The clock design supports several methods for setting and adjusting time. Both the Spacecraft Time and UTCF can be 'Jam Load' commanded to a specific value. Also, the UTCF can be commanded to 'Delta Adjust' by a specific number of subseconds. This can be commanded to have a one-time effect or can be commanded to adjust the UTCF every 1 Hz cycle. There is no plan to use the 1 Hz delta adjust capability.

4.1.7.3 Spacecraft Time Correlation

The mission requires accurate spacecraft time correlation with the ground for science data processing and for nominal operation of several on-board software functions including packet time-tagging, stored command execution, and on-board attitude computations. The FDS design supports two time correlation methods, the Range Data Delay (RDD) method and the User Spacecraft Clock Calibration System (USCCS) method. Ground data system software (MOC and WSC) also support both time correlation methods. Section 6 describes operations relating to time maintenance.

4.1.8 On-Board Recorder Operations

The TRMM FDS design includes a Solid State Recorder (SSR) for storing spacecraft and instrument housekeeping, science data, and for logging event messages generated by the flight software. An area of bulk memory along with the required flight software constitute TRMM's Solid State Recorder (SSR). The SSR bulk memory consists of 14 memory cards, each containing 20 MB for a total storage of 280 MB. This is equivalent to approximately two orbits of record capacity.

The Data Storage (DS) task of the flight software provides management of all data recording. Figure 4.1-16 illustrates the DS task, system interfaces, and on-board data flow.

The Data Storage task receives data packets from the various instrument and spacecraft sources across the S/C processor's software bus, formats these packets into CCSDS Advanced Orbiting Systems (AOS) Virtual Channel Data Units (VCDUs), and stores the VCDUs in the SSR bulk memory.

The mission operations plan is to operate all recorders (VR 1-6) in the Normal Playback mode. In addition, playback of all recorders (science and housekeeping) will occur during all TDRS events. The Flight Status Message recorder (VR7) or Event Buffer will be dumped during all TDRS events and will be telemetered to the ground on VC 0.

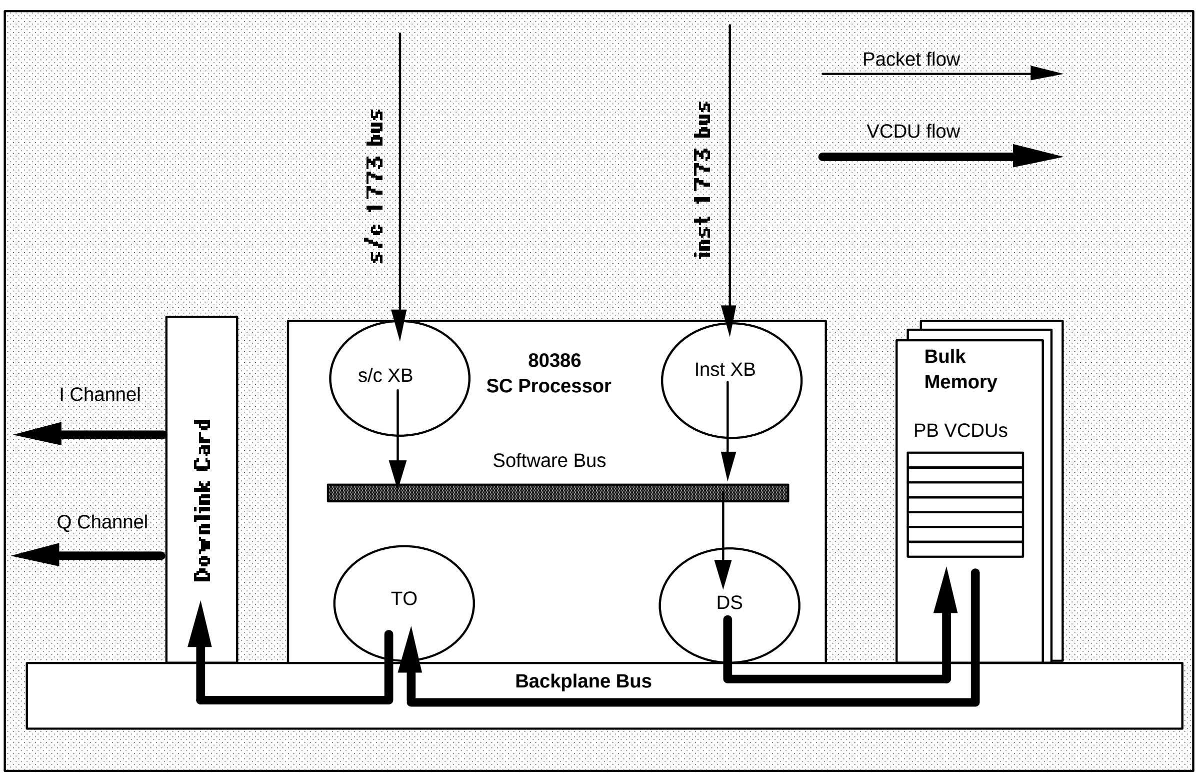


Figure 4.1-16 DS Task and System Data Flow

4.1.8.1 SSR Bulk Memory Management

The SSR is partitioned into seven Virtual Recorders (VRs), each providing memory allocation for a specific user. Specific operations regarding VR7 containing spacecraft event messages are found in Section 4.1.8.4. Table 4.1-10 reflects VR assignments and operations modes.

VR ID	Data Type	Downlink VC	Retrans VC	Playback Mode	Record Mode
1	Housekeeping Data	VC 1	VC 11	Normal	Non-Overwrite
2	CERES Data	VC 2	VC 12	Normal	Non-Overwrite
3	LIS Data	VC 3	VC 13	Normal	Non-Overwrite
4	PR Data	VC 4	VC 14	Normal	Non-Overwrite
5	TMI Data	VC 5	VC 15	Normal	Non-Overwrite
6	VIRS Data	VC 6	VC 16	Normal	Non-Overwrite
7	Spacecraft Event Messages	VC 0	N/A	Rec. Dump	Non-Overwrite

Table 4.1-10 TRMM VR Assignments and Operating Modes

The SSR design provides the capability to modify bulk memory allocation of individual VRs in-flight. The total allocation (per VR) plus any unassigned memory may not exceed the total SSR bulk memory size. Also, the allocation of a VR may not be reduced below the memory currently in use by the VR. An operations scenario for modifying memory allocations follows:

a. Command Playback and Data Release of the VR from which memory is to be taken.
b. Command a reduction of the memory allocation of that VR thereby increasing a "Free Memory Pool".
c. Command an increase to the allocation of the VR up to the memory available in the Free Memory Pool.

Prior to launch, the size of VR 1 will be increased allowing up to 30 hours of spacecraft housekeeping and engineering data to be stored. Upon completing functional checkout of the spacecraft subsystems, and prior to initiating normal 'Science' operations, all VR memory allocation will be configured to the nominal mission assignments. During nominal operations, VR allocations will not be changed. VR allocations may be changed in the event of an instrument failure.

4.1.8.2 SSR Data Storage

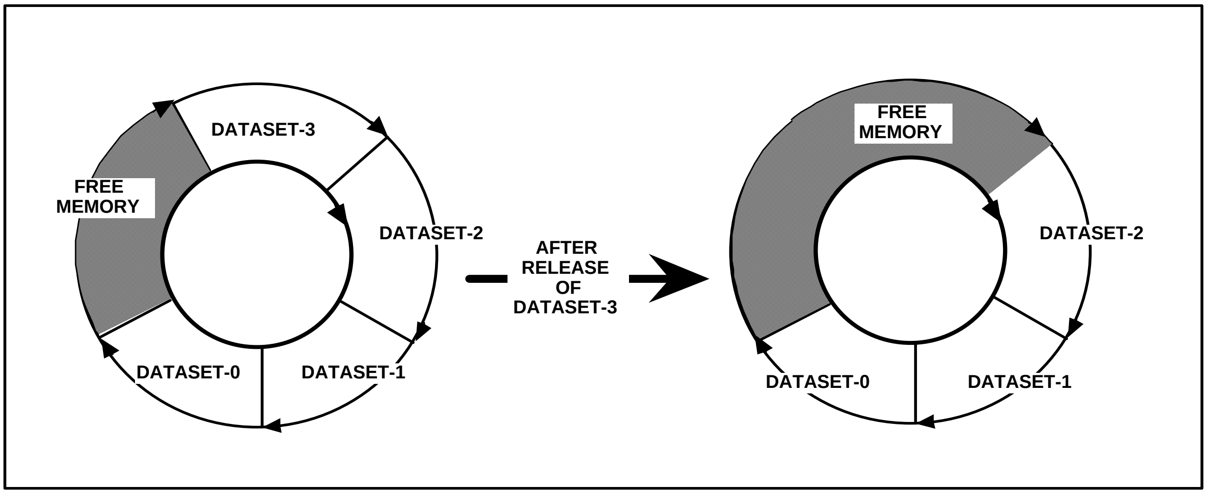
Recorded data in each VR is segmented into partitions. Operationally, these partitions are called datasets. The most current data is always recorded into Dataset-0, with up to three additional datasets supported per VR.

Housekeeping information in telemetry is provided indicating the oldest VCDU sequence number recorded in each of the four datasets and is included for each VR. Other telemetered housekeeping information includes the latest recorded VCDU sequence number, and the total free memory blocks. From this information the MOC will determine which VCDUs are in each dataset and how much data to expect for each VR playback, as well as how much memory is still available per VR.

The SSR DS task supports Non-Overwrite and Overwrite record modes. When a VR is operated in the Non-Overwrite mode, data recording to a particular VR will discontinue when its capacity has been reached. Data will not be overwritten. To free space and allow record operations to continue, bulk memory must be played back and then released by ground command in this mode. When a VR is operated in the Overwrite mode, logical blocks (containing 16 VCDUs) of bulk memory are automatically released (essentially lost if not already played back), by the DS task once a VR's memory allocation has been reached. For normal mission operations, all VRs will be operated in the Non-Overwrite mode.

Recorded VCDUs may be released by specified dataset or by sequence number. When a given dataset is released all older datasets are released. For example, in a scenario where three datasets exist, and a command to release Dataset-1 is issued, Datasets-2 and -3 will also be released.

Each frame is assigned a unique sequence number, per VR, as it is recorded. When data is released by sequence number, all VCDUs with sequence numbers less than the specified sequence number are released. For example, when releasing frame #100, all older frames are released, regardless of what dataset they reside in. Figure 4.1-17 illustrates the VR dataset and release concept.



4.1.8.2.1 Packet Filtering and Storage Quotas

The DS task provides the capability to record data at intervals lower than the actual data acquisition rate. This feature is referred to as packet filtering. Filtering is based on CCSDS Applications Process ID (APID) and allows "n of x" packets to be recorded. For example, if a particular APID is filtered by (3,6), DS records the packet if its packet sequence number mod-6 is less than 3. In this example, three of every six packets would be stored in the recorder. Table 4.1-11 illustrates which packets would be recorded in this example.

Packet Sequence Number	Calculation	Less than three?	Result
0	Mod (0,6) = 0	Yes	Stored
1	Mod (1,6) = 1	Yes	Stored
2	Mod (2,6) = 2	Yes	Stored
3	Mod (3,6) = 3	No	Discarded
4	Mod (4,6) = 4	No	Discarded
5	Mod (5,6) = 5	No	Discarded
6	Mod (6,6) = 0	Yes	Stored
7	Mod (7,6) = 1	Yes	Stored

Table 4.1-11 Example (3,6) DS Filter Logic

In addition to the packet filtering, the DS task controls data storage by employing storage quotas. This prevents any single on-board data source from monopolizing an entire VR. Specific packet filtering, APID quotas, and specific APID assignments to a particular VR will be controlled by on-board system tables. These tables will be defined prelaunch for TRMM and may be modified in flight to accommodate specific contingencies.

4.1.8.3 SSR Playback Operations

During normal operations, the FOT will not command recorder playbacks in real-time. Instead, recorder playbacks will be controlled through RTSs contained in the daily command load. Each recorder will be closed prior to being played back.

Playback transmission rates are specified in conjunction with the real-time rates in an on-board system table referred to as the Rate Index Table. All recorder playbacks will be telemetered on the TDRS Q-Channel. Table 4.1-11 reflects TRMM's current Rate Index Table definition.

Rate Index #	I Rate (Kbps)	Q Rate (Kbps)	Real-time Channel	Purpose
1	1 (Fill)	1	Q	OMNI/SafeHold. Fill packets on I-Channel
2	1 (Fill)	1.5	Q	Omni/Event Message and Memory dump operations
3	32	128	I	HGA/Center Frequency Measurements. and HGA/Coarse HGA pointing Operations.
4	32	2048	I	HGA/Normal Operations
5	N/A	1	Q	GSTDN/Real-time Only
6	N/A	1024	Q	GSTDN/Recorder Playback
7	1 (Fill)	1	Q	Spare (Set to Default)
8	1 (Fill)	1	Q	Spare (Set to Default)
9	1 (Fill)	1	Q	Spare (Set to Default)
10	1 (Fill)	1	Q	Spare (Set to Default)

Table 4.1-12 TRMM Rate Index Table

4.1.8.3.1 SSR Playback Modes

The SSR design supports three modes, a Normal Playback mode, a Continuous Playback mode, and a Recorder Dump mode. The following paragraphs describe each mode in more detail and identifies which TRMM VRs will be operated in each mode.

VRs may be operated in the Normal Playback mode on an individual basis. When recorder transmission is commanded in the Normal Playback mode, Dataset-0 is closed and renamed as Dataset-1. A new Dataset-0 for continued recording is created. VCDUs are transmitted to the ground beginning with the oldest VCDU in Dataset-1. At the point when all recorded VCDUs have been transmitted to the ground in Dataset-1, playback from that VR stops. Playback of Data Set 1 (VR 1 - 6) will routinely occur every orbit during scheduled TDRS SSA events.

VRs may be operated in the Continuous Playback mode on an individual basis. When recorder transmission is commanded in the Continuous Playback mode, the TO task begins to transmit VCDUs to the ground beginning with the oldest VCDU in Dataset-0. At the point where all older recorded VCDUs have been transmitted to the ground, new data will continue to be written to Dataset-0, and then will immediately be transmitted to the ground effectively providing near real-time downlink. Figure 4.1-18 illustrates the Continuous Playback mode.

Upon receipt of a Playback_Off command, Dataset-0 will be closed at the point of the last transmitted VCDU. FDS tasks then reset pointers and rename the dataset as Dataset-1. All VCDUs continue to be stored in a new Dataset-0.

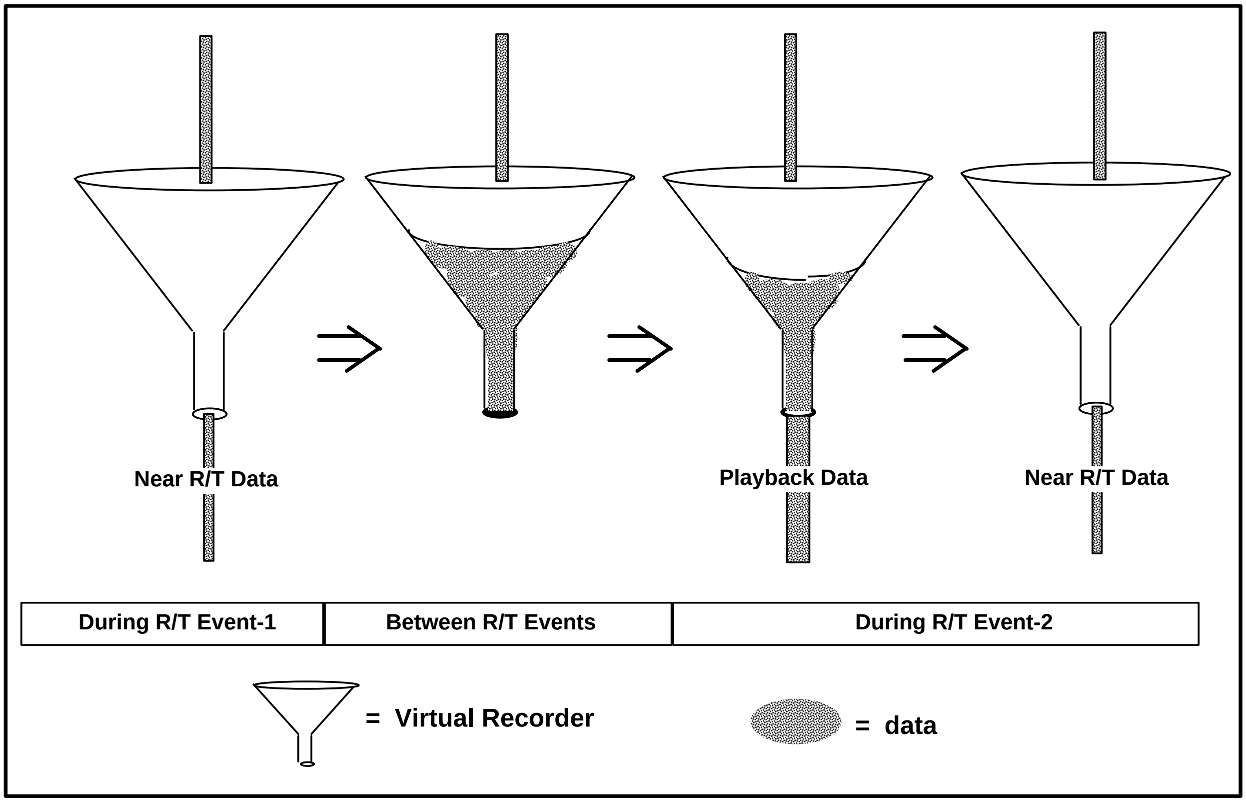


Figure 4.1-18 Continuous Playback Concept

The Recorder Dump mode is designed specifically to allow Flight Status Messages stored in VR 7 (Event Buffer) to be downlinked on the real-time housekeeping virtual channel, thereby allowing real-time processing by the MOC. When VR 7 transmission is commanded in the Dump mode, handshaking between the DS and TO tasks close Dataset-0 renaming it to Dataset-1, and a new Dataset-0 is opened for continued Flight Status Message recording. The TO task then reads VCDUs from VR 7 beginning with the oldest VCDU in Dataset-1. These packets are then processed along with all other housekeeping packets destined for real-time downlink on VC 0. Only VR 7 (the Event Buffer) is planned to be operated in this mode.

4.1.8.4 SSR Retransmission Options

The SSR design supports various methods of replaying recorded data to the ground. One method is to command a second playback of the VR. This method downlinks the data on the same VC and is processed identically by the MOC and SDPF.

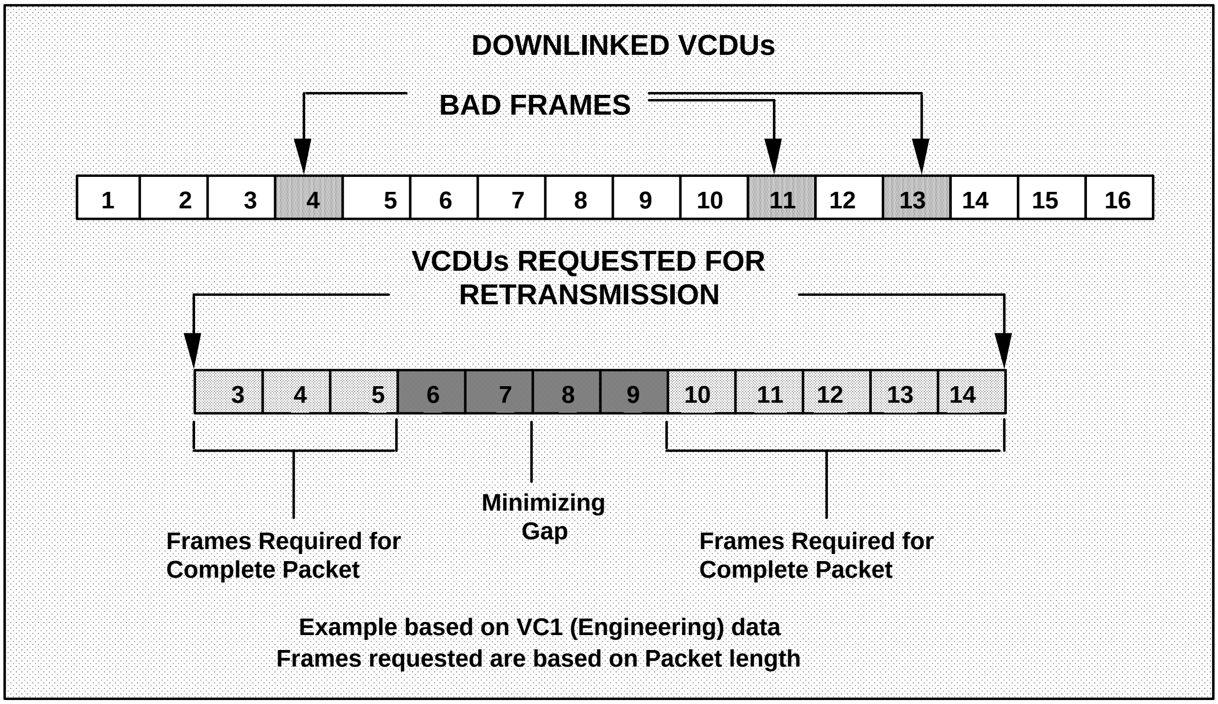
Another method is to command a retransmission of lost or corrupted data to the ground. An advantage in retransmission is that specific VCDUs may be transmitted, rather than entire datasets. This reduces the amount of data transmission to the ground and increases the probability of capturing all recorded data. Retransmitted VCDUs are downlinked on an alternate VC, as previously listed in Table 4.1-8.

The retransmission command specifies a sequence of frames to be replayed, and up to 30 sequences may be specified per command. Upon receipt of a retransmission request command, one sequence will be processed, but the next sequence will not be processed until the next second. This implies a minimum of 30 seconds to process a command containing 30 VCDUs. There is no buffering of retransmission commands. The MOC will use end-item telemetry verifiers to delay transmission of retransmission commands appropriately. The downlink "frame rate" is 200.16 "frames" per second at 2 Mbps.

For each retransmitted sequence, the replay flag will be set for the first VCDU transmitted. This will be used as an indicator for Pacor when performing Level-0 processing on the data.

When specifying frames for retransmission, a complete packet must be retransmitted to ensure correct processing on the ground. Since science packets span multiple frames, a range of frames, dependent on the VR, will be requested for each missing frame. The MOC will employ an algorithm to efficiently determine which frames are to be retransmitted. Figure 4.1-19 illustrates requested frames against missing frames, using the efficiency algorithm. The MOC algorithm will:

a. Request frames on either side of the missing frame to recreate one packet. Requested frames will be based on the packet length per VC.
b. Provide for a gap between frames, which will minimize the number of requested sequences.


DOWNLINKED VCDUs
BAD FRAMES
1 2 3 4 5 6 7 8 9 10 11 12 13 14 15 16
VCDUs REQUESTED FOR RETRANSMISSION
3 4 5 6 7 8 9 10 11 12 13 14
Frames Required for Complete Packet
Minimizing Gap
Frames Required for Complete Packet
Example based on VC1 (Engineering) data
Frames requested are based on Packet length

4.1.8.5 Event Buffer Operations

Flight Status Messages are generated on-board TRMM in response to a variety of error conditions. These messages can originate from the ACE, ACS, and S/C processors and are formatted as individual data packets with unique APIDs to identify the message source. Each Flight Status Message is included in the real-time downlink on VC 0 when the message is generated during a real-time contact. Event messages generated between real-time supports are recorded in the Event Buffer. The Event Buffer has been assigned to VR 7, therefore, operation and control capabilities of the Event Buffer will be similar to those capabilities supported for all VRs. However, the Event Buffer will be dumped on VC 0 and multiplexed into the real-time telemetry stream. In this way, Flight Status Messages generated between TDRS contacts can be downlinked during the subsequent contact and processed immediately by the MOC real-time system.

4.2 ATTITUDE CONTROL SUBSYSTEM

The Attitude Control Subsystem (ACS) provides three-axis stabilization and controls vehicle maneuvers to satisfy the mission requirements. TRMM is a nadir pointing, three axis stabilized spacecraft. The X axis (plus or minus) points in the velocity vector. A complement of an Earth Sensor Assembly (ESA), Inertial Reference Unit (IRU), and Digital Sun Sensors (DSS) provide attitude information. Coarse Sun Sensors (CSSs) are used to provide attitude information during SafeHold and Sun Acquisition modes. The Earth's magnetic field is sensed by a Three Axis Magnetometer (TAM).

Control torques will be provided by Reaction Wheel Assemblies (RWAs), with excess momentum unloaded via Magnetic Torquer Bars (MTBs) reacting against the Earth's magnetic field. An Engine Valve Driver (EVD) is used by the ACS to control the Reaction Control Subsystem (RCS) Thrusters.

Several ACS attitude determination and control algorithms reside in the ACS processor. Each determination algorithm employs a combination of sensors to compute euler angles, body rates or transformation matrices between inertial and body coordinates. Each attitude control mode uses a combination of the determination algorithms to develop commands to be delivered to the Reaction Wheels. In addition to attitude determination and control, the ACS software provides ephemeris propagation and control functions in support of Solar Array (SA) and High Gain Antenna (HGA) pointing. A Failure Detection and Correction (FDC) function which monitors ACS components is also provided. SafeHold mode logic processing is located separately in the Attitude Control Electronics (ACE) that can independently determine attitude and command the actuators in the event of an ACS failure.

A number of science objectives translate into orbit and attitude requirements. The ACS will maintain pointing knowledge to within 0.18° per axis and pointing control to within 0.32° per axis. Pointing stability is kept to within 0.1° over one second, per axis. The PR instrument requires that the geodetic altitude (350 km reference altitude) be maintained to within +7/-8 km.

Project requirements state that the altitude be maintained to ±1.25 km, which will accomplish the PR requirement. The VIRS instrument cooler (located on the +Y side of the spacecraft) must not be pointed towards the Sun, which requires the spacecraft to perform a 180° yaw turn every two to four weeks, as orbit regression and Sun motion causes the Sun to rise above or go below the orbit plane. The CERES instrument requires a Deep Space Calibration, which the ACS meets with an inertially fixed attitude configuration. The PR instrument requires a -Y forward orientation for Antenna Pattern Measurements.

4.2.1 Component Description

The ACS is made up of a number of components. They include the ACS, ACE and GSACE Processors, as well as the Actuators and Attitude Sensors. The Actuators are the MTB, the RWA and the EVD. The attitude Sensors are the CSS, DSS, the ESA, the IRU and the TAM.

Connected to the GSACE are the Deployables. The deployables are the HGA and the SA. The general connectivity of the entire system is shown in figure 4.2-1.

Subsequent discussion will focus on those components that belong to the ACS System. This includes the ACS, ACE and GSACE processors, as well as the attitude sensors and actuators. Rocket Engine Modules (REMs) and other components of the RCS are described in section 4.7. Deployables are described in section 4.8.

Table 4.2-1 summarizes the ACS components:

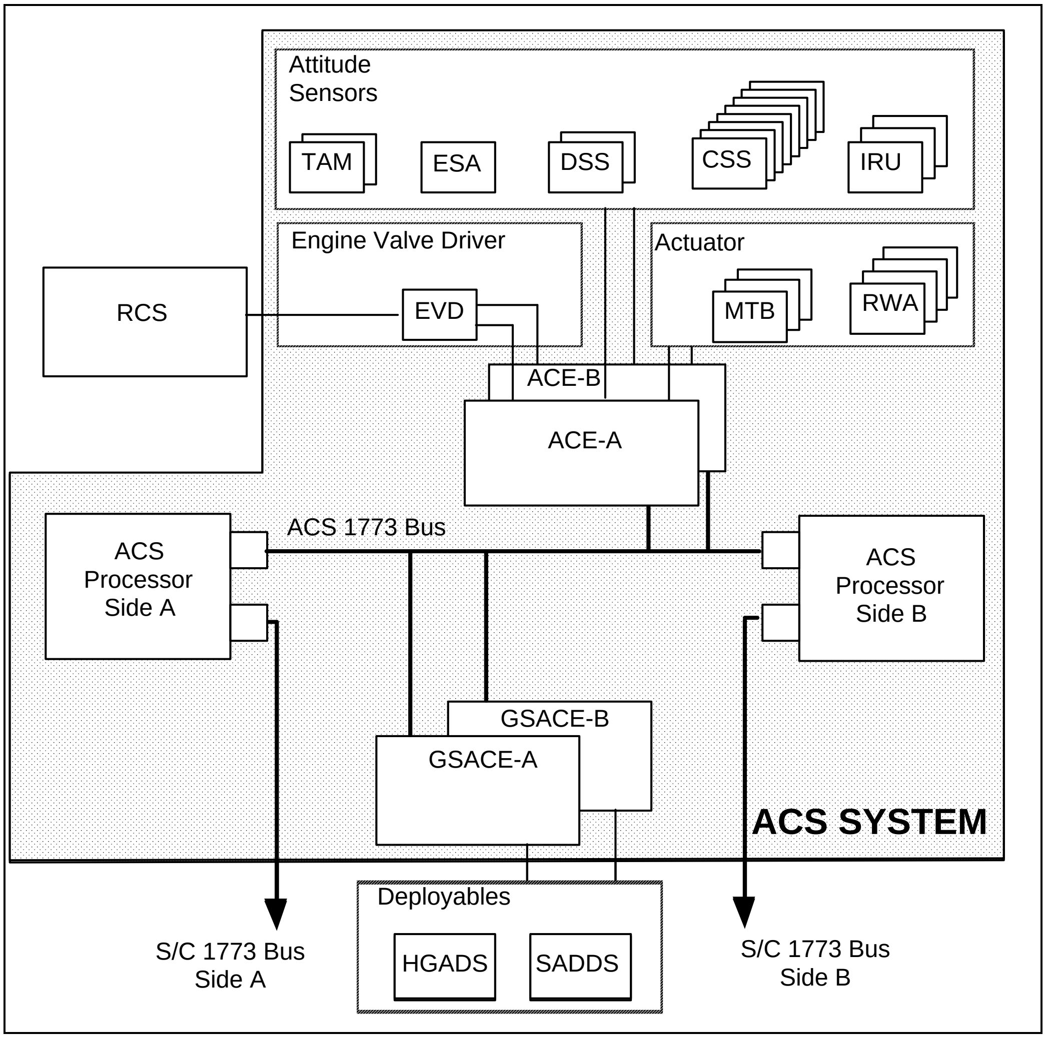


Figure 4.2-1 ACS Block Diagram

Component	Acronym	Number	# Required For Mission Objectives	Location
Attitude Control Electronics	ACE	1 unit, internally redundant	1	LBS
Engine/Valve Driver	EVD	1 unit, internally redundant	1	LBS
Earth Sensor Assembly	ESA	1 unit, internally redundant, w/ 4 sensors	1, w/ 3 sensors	LISP
Inertial Reference Unit	IRU	3	2	ISP
Digital Sun Sensor	DSS	2	1	Elec: LBS Heads: LBS, UISP
Coarse Sun Sensor	CSS	8	4	SA, UISP
Three Axis Magnetometer	TAM	2	1	UISP
Reaction Wheel Assembly	RWA	4	3	ISP
Magnetic Torquer Bar	MTB	3, 2 windings per MTB	3,1 winding per MTB	LBS

Table 4.2-1 ACS Component Summary

4.2.1.1 ACS Processor

The ACS processor is a 80386 16 MHz microprocessor with a 80387 coprocessor. It provides the platform on which the ACS control algorithms reside. The ACS processor is connected to two 1773 data buses. These are the Spacecraft Bus and the ACS Bus. The Spacecraft bus connects the ACS to the SC Processor. The ACS is a Remote Terminal (RT) on that bus. The ACS Bus connects the ACS Processor to the GSACE Processor, the ACE Processors, as well as the redundant ACS Processor. The ACS processor is designated as the Bus Controller (BC) for this bus.

<u>Tasks Running on the ACS Processor</u>
Several processes run on the ACS Processor. Many of them are the same as those running on the SC Processor, such as the Software Manager, as well as a Stored Command Processor Task. A full description of the SDS architecture and FDS operation on the ACS processor is provided in section 4.1.

<u>The ACS Task</u>
In addition to the above tasks, the ACS Processor also runs an ACS task. This is the key task of the ACS Processor. It is this task that employs attitude data, propagates the current TRMM, TDRS and COMETS positions, determines what actuator response is needed to maintain the correct attitude, commands the actuators, commands the deployables, and finally, conducts detection and correction of ACS system failures.

Two Modes of Operation for the ACS Task

The ACS task operates in either a 2 Hz or 8 Hz mode. The 2 Hz mode is the normal operational mode and provides attitude control for science collection. The 8 Hz mode operates during either the Delta-V or Delta-H thruster modes (described below). For both 2 Hz and 8 Hz modes, the execution of each ACS control cycle is timed to the receipt of ACE 2 Hz or 8 Hz telemetry data.

A detailed description of the functions performed by the ACS task is described in the TRMM ACS Flight Software User's Guide [TRMM-712-184]. A description of the control modes is also provided later in this section.

4.2.1.2 Attitude Control Electronics

This section will limit description to only the key elements of the ACE that relate to operations. Full details of the ACE function can be found in the TRMM ACE Software User's Guide [TRMM-712-183].

Two Main Functions
The Attitude Control Electronics (ACE) provides two main functions. The first is the interface between the sensors/actuators and the ACS processor. The ACS sends commands to the ACE over the 1773 bus. The ACE interprets these commands and generates signals for driving the actuators. It also takes the signals from the attitude sensors, converts them into digital data, and sends this information in packets to the ACS over the 1773 bus.

The second function of the ACE is to monitor the health of the ACS. In the event that the ACS fails, the ACE will automatically take over. The ACE will notify the other spacecraft subsystems over hard wire lines. It will also take control of the spacecraft attitude, by maintaining a SafeHold configuration.

Architecture
The ACE box is a single, internally redundant unit with ACE A designated as the primary control unit and ACE B as the redundant unit. Each ACE is cross-strapped to each ACS sensor and actuator, except for the CSSs and MTBs. In normal operations, both ACE A and B are powered ON, with ACE A selected for actuator control. Telemetry is provided from both ACEs.

Each ACE consists of a microprocessor/memory board operating at 1.5 MHz, a number of interface cards, and low voltage power supplies. The following paragraphs provide a short description of internal ACE components.

a. A Harris 80C85 Microprocessor/Memory board, operating at 1.5 MHz and containing 32 KB EEPROM, 2 KB ROM and 62 KB RAM. The processor/memory board provides digital SafeHold logic, a 1773 fiber optic interface, synchronized timing signals for ACE circuits, and interface and control of ACE I/O boards. There is no coprocessor or floating point arithmetic support available.
b. An Actuator card, providing an interface for RWA commands and MTB drivers. The actuator board provides the bi-level SafeHold signal to each GSACE and Instrument PSDU.
c. A Torquer Bar Driver card, connecting to one winding of each MTB unit. Torquer bar driver boards are not cross-strapped, with ACE A controlling winding A and ACE B controlling winding B.
d. A Digital Sun Sensor/Earth Sensor card, providing an interface to the two DSS units and the ESA electronics. The DSS provides Sun azimuth, elevation, and presence information.

As interface to the ESA electronics, the card receives ESA data and provides the 2 Hz synchronizing pulse and data clock to the ESA.

e. An IRU board, interfacing with the IRU to receive 6 channels of incremental angle pulses, which are provided to the microprocessor at 8 Hz.
f. A Multiplexer card, interfacing with all ACS sensors and actuators and digitizes the analog signals used for telemetry and spacecraft control.
g. An Engine/Valve Driver card, interfacing with EVD A and EVD B.
h. Two Low Voltage Power Supplies, each cross-strapped to IPSDU A and IPSDU B.

4.2.1.3 Earth Sensor Assembly

The static ESA, supplied by Barnes Engineering, is the primary attitude sensor providing roll and pitch information. The interface with the ESA will be described first. Then there will be a discussion of the principles of operation of the ESA, including how it will handle Sun or Moon interference, calibration parameters and the possibility of fogging of the ESA detectors and calibration parameters that can be adjusted to account for it.

<u>ESA Interface with the ACE</u>
The ESA is a single unit with redundant processing electronics and no moving parts. The electronics are cross-strapped to ACE A and ACE B. Nominally, ESA A electronics are connected to ACE A and ESA B electronics are connected to ACE B. The spacecraft will be capable of operating with any three functional detectors. Figure 4.2-2 shows the ESA-to-ACE interfaces.

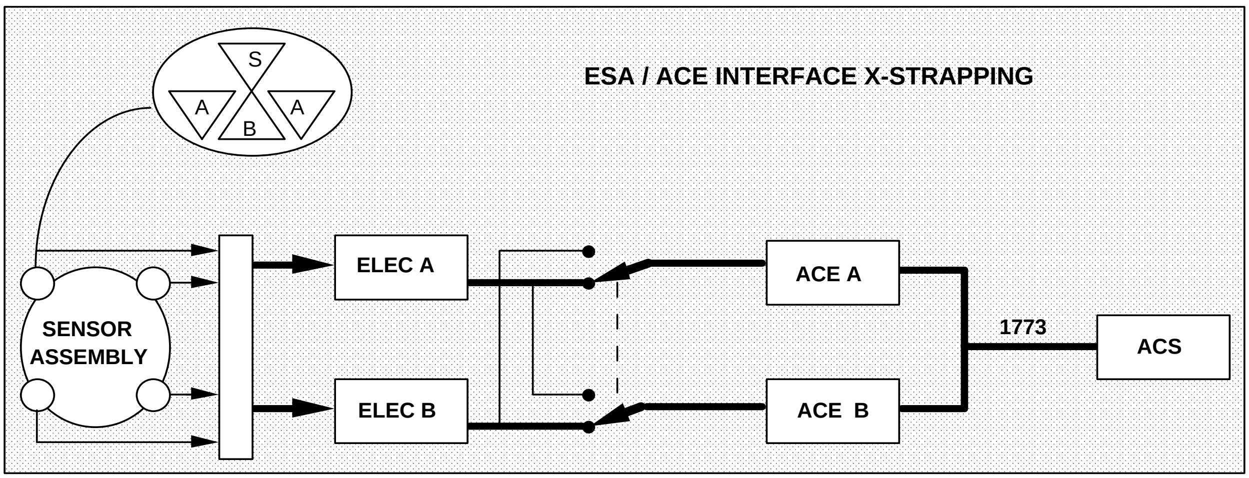


Figure 4.2-2 ESA/ACE Interfaces

The ESA has an operating altitude of 335 to 390 km, with degraded performance down to 200 km. The ESA provides an attitude error of less than 0.08° ±7%, and has the capability of determining Sun/Moon interference.

<u>Principle of Operation</u>
The ESA operates by sensing the contrast between the cold of deep space and the heat of the Earth's horizon, which is imaged on four internal infrared thermopiles located at 90° spacing.

The depression angle x, shown in figure 4.2-3, is calculated for each of the four detector quadrants and from the four depression angles, pitch and roll information can be derived. If all four detectors are available, and the direction of flight is as shown, the equations are:

$$\boxed{Pitch = \frac{\sqrt{2}}{4}(x_2 - x_4 + x_3 - x_1)} \quad \text{and} \quad \boxed{Roll = \frac{\sqrt{2}}{4}(x_2 - x_3 + x_4 - x_1)}$$

There are two modes of operations that utilized the ESA data, Coarse ESA acquisition and Fine ESA acquisition. Coarse is used during Earth Acquisition mode. Once Earth Acquisition is successful, the ESA processing switches to Fine mode. Fine mode is used in Yaw Acquisition mode and Nominal Mission mode. Coarse mode will be again be used for end-of-life operations.

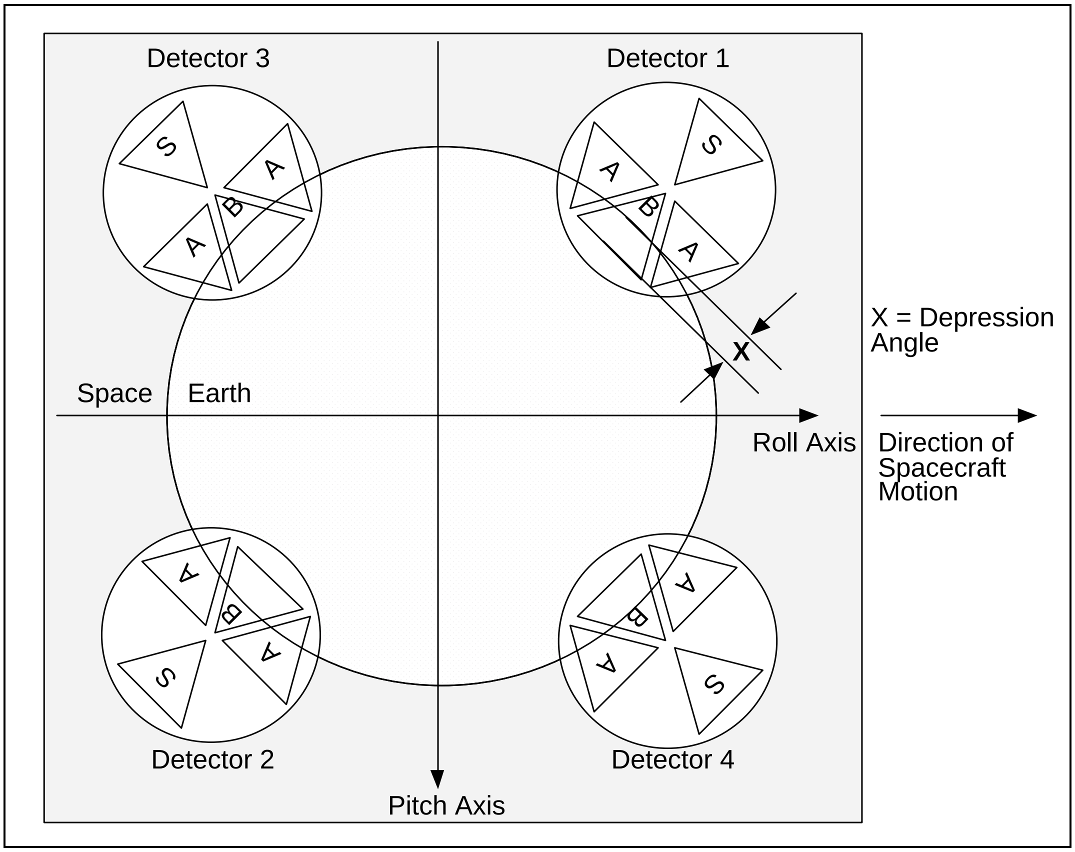


Figure 4.2-3 ESA Detectors and Their Orientation in Flight

Both Coarse and Fine ESA modes will compute the depression angle. The Fine mode improves the calculation by filtering irradiance and employing the S head in each detector.

Sun and Moon Interference

In the Fine mode, if the Sun or the Moon comes within 10.25° of a given detector, it will not be used as part of the algorithm. In this case, one can use that $x_1 + x_2 \approx x_3 + x_4$. For example if the Sun were too close to detector one then $x_1 \approx x_3 + x_4 - x_2$. This can be plugged into the above equations to get Pitch and Roll in terms of the remaining depression angles.

ESA Calibration Variables and Fogging

In flight, the FDF will examine ESA performance. Based on this, the FDF will provide calibration information which the MOC will upload to the spacecraft. These calibrations will update values in ACS tables 59 and 60.

The calibration information will be:

- Detector fine bias (one for each detector A, B or S, in each quadrant 1, 2, 3 or 4)
- Penetration angle scale factor (one per quadrant)
- Penetration angle bias (one per quadrant)

- Sensor alignment matrix (body to true alignment, including misalignments)

The key entries within table 59 are in the matrices ESAFINK, ESAPENK and ESAPENC. The parameters in table 60 are in the matrix ESA2BDY.

Previously flown ESAs of the same manufacture have demonstrated a fogging of the optics due to atomic oxygen encountered in the space environment. Operationally, this translates into an increased number of expected calibrations and close monitoring of ESA operations. Strategies for adjusting the table parameters are still preliminary, but generally speaking, one can adjust ESAFINK and ESAPENK to account for fogging of the ESA optics. ESA2BDY is the transformation matrix that accounts for sensor alignments and misalignments in the mounting on the spacecraft body

4.2.1.4 Inertial Reference Unit

The IRU senses the spacecraft's angular rates. TRMM uses a single IRU unit equipped with three two axis gyros. The IRU is used for rate damping for all three spacecraft axes and for attitude information along the axes.

<u>Redundancy</u>
The IRU is configured to provided redundant outputs for the X, Y, Z body axes. Each two axis gyro senses motion about two input axes: (X, Y), (X, Z), and (Y, Z), as shown in figure 4.2-4:

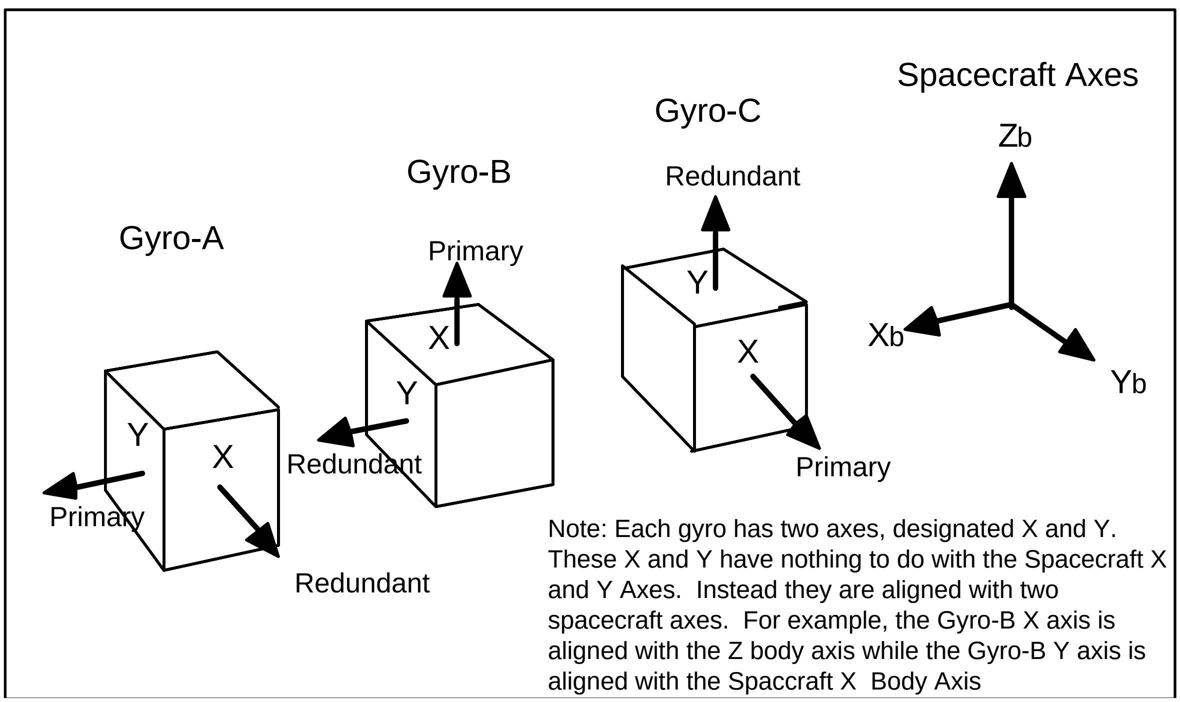


Figure 4.2-4 IRU Channel Orientation Along Spacecraft Axes

The choice of whether to use the primary or redundant IRU for a given spacecraft axis is commandable. This command can originate either as an FDC response or a ground command.

High and Low Rate Modes
The IRU may operate in either the High Rate Mode and the Low Rate Mode. The High Rate Mode is meant to be used to measure the angular rate of the observatory when it is rotating quickly (at a high rate). This mode will allow the IRU to make measurements within an environment of up to ±4.0°/second. Above this rate, the IRU will be saturated. It will not be able to make as accurate a measurement as the Low Rate Mode. The Low Rate Mode is meant for slow rotation rates (at a low rate). In this mode the IRU can make measurements in a rotating environment of up to ±800 arc-sec per second. TRMM however will only use the High Rate Mode.

IRU Measurements
The IRU puts out two kinds of information. They are the Incremental Count and the Analog Rate. The Analog Rate is a direct measure of how many degrees per second of spacecraft rotation that each axis of the IRU measures. The analog value, however, is linear only for rates between ±2.0 degrees per second.

The Incremental count is a digital signal. The IRU provides a pulse every time the spacecraft angular position changes by a certain amount around a given axis of the IRU. In the High Rate Mode, there are 1.6 arc-sec per pulse while in the Low Rate Mode there are 0.1 arc-sec per pulse. Note that there is a trade off here. The high rate mode can provide accurate information for a wider range of angular rates of the spacecraft, but with lower resolution than the low rate mode.

4.2.1.5 Digital Sun Sensors

There are two Adcole DSSs on board TRMM, providing Sun presence, elevation, and azimuth information used for yaw reference. Each DSS has two one-axis sensor heads as well as electronics. The heads are orthogonally mounted, and have a 100 x 100° field of view. The DSS can detect attitude errors to 0.05°. One DSS units is placed fore (+X) and one aft (-X) for Sun detection in either spacecraft orientation. Each DSS is cross-strapped to each ACE. Figure 4.2-5 shows DSS/ACE interfaces.

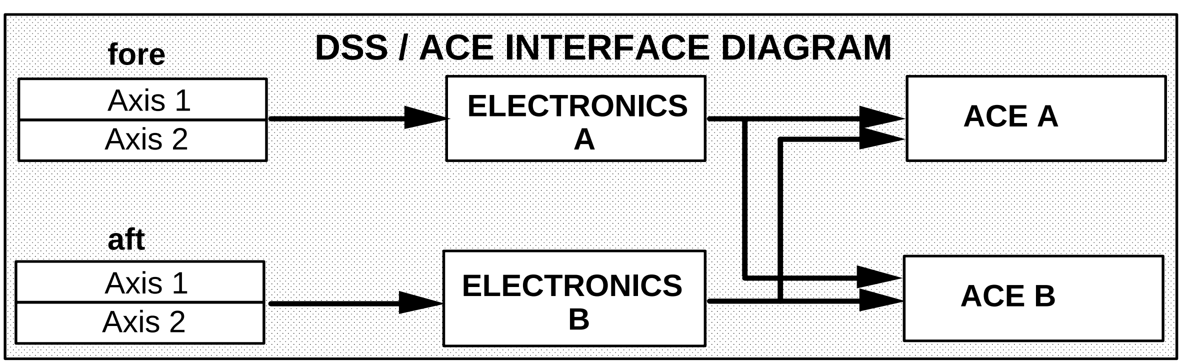


Figure 4.2-5 DSS/ACE Interface Diagram

DSS are Used for Yaw Updates

The IRU is used to determine the yaw of the spacecraft. The IRU's themselves measure yaw rate. To get yaw angle, the ACS integrates the yaw rate for a period of time. The uncertainty in the measurement of the yaw rate will show up as an accumulated error in the yaw angle calculated. This error can be minimized by estimating the gyro drift and the gyro error angle. The gyro drift compensates for the errors in the measured gyro rate while the error angle looks the accumulated error in the integration process. When TRMM is nadir pointing, both the fore and aft DSS will see the sun during the orbit. The sun will be in view of one at dawn, and the other at dusk. So the yaw updates are applied at dawn and at dusk when TRMM is assured to have valid Sun information. If a DSS were to fail, the update will only be performed once per orbit, when the remaining DSS can see the sun. Analysis has shown that the mission requirement for attitude determination can be achieved with only one yaw update per orbit.

The ACE provides a "valid DSS" data flag. It is passed to the ACS processor. If the DSS is OFF, stops transmitting, or has not updated since the last microprocessor read, the data flag is set. The ACS takes 20 samples of DSS data. Once ten good samples or ten bad samples are collected, no further samples are taken. If ten good samples were obtained, these values are averaged and used to perform the Yaw update. If ten good samples were not obtained, or the DSS output does not compare with the ephemeris, the DSS is marked as questionable and the Yaw update is not performed.

4.2.1.6 Coarse Sun Sensors

Eight Adcole CSSs are on board TRMM, grouped in two sets of four sensors. Each set of four units provides information to either ACE A or ACE B. Each CSS is a photovoltaic eye, which is used in combination with the other eyes to provide a Sun reference angle to within 10°. Each eye has a 170° field of view, so a set of four eyes can provide a 4π steradian Sun coverage. CSS inputs are only used in the SafeHold and Sun Acquisition control modes. CSS signals to each ACE are made available to the other ACE, providing the first ACE is powered ON. Figure 4.2-6 provides an illustration of the ACE/CSS interfaces.

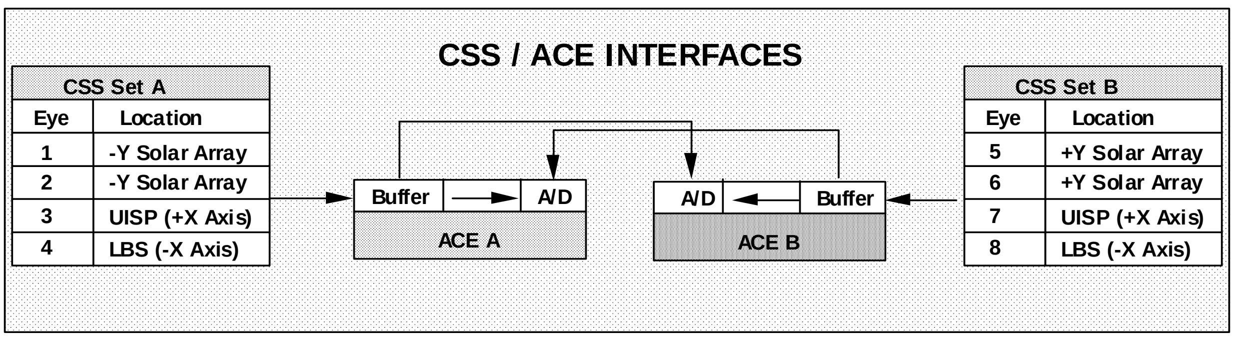


Figure 4.2-6 ACE/CSS Interface Diagram

4.2.1.7 Three Axis Magnetometers

Two TAMs sense the direction and strength of the Earth's magnetic field. This information, along with the reaction wheel speeds and gyro data will be used to drive the magnetic torquer bars to react against the Earth's magnetic field to unload excess system momentum. Each magnetometer senses magnetic fields up to 60 microTesla (600 milligauss) with a scale factor of 16.6 millivolts per milligauss. The TAM is fully redundant, with electronics cross-strapped to the ACE, as shown in Figure 4.2-7.

4.2.1.8 Reaction Wheel Assembly

Four Ithaco RWAs, arranged in the pyramidal configuration shown in figure 4.2-8, provide the control torques necessary to maintain the desired spacecraft attitude in the presence of disturbance torques. During Mission mode, a torque distribution algorithm will be employed to equalize and minimize wheel speeds. The RWA pyramidal arrangement allows for fault tolerance and maximizing the momentum storage along a preferred axis. Attitude control requirements may be met by any three wheels.

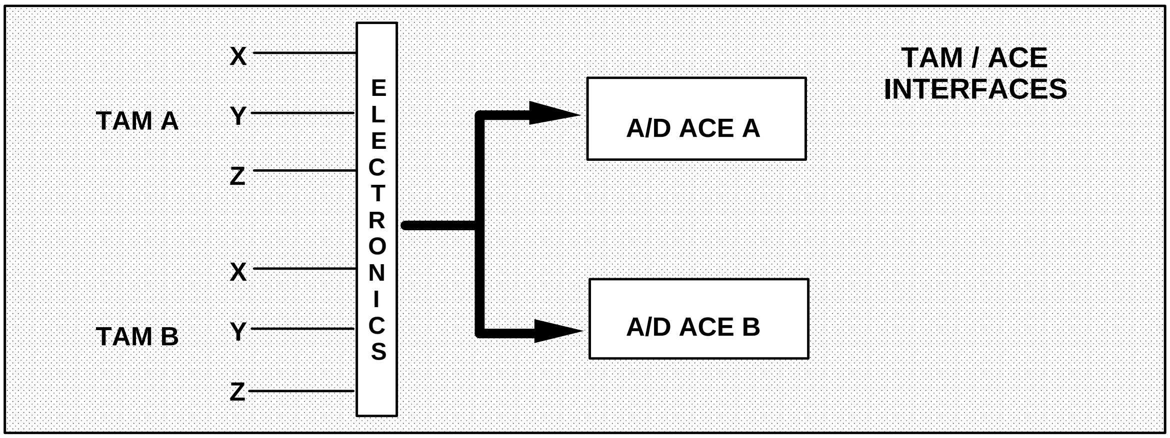


Figure 4.2-7 ACE/TAM Interface Diagram

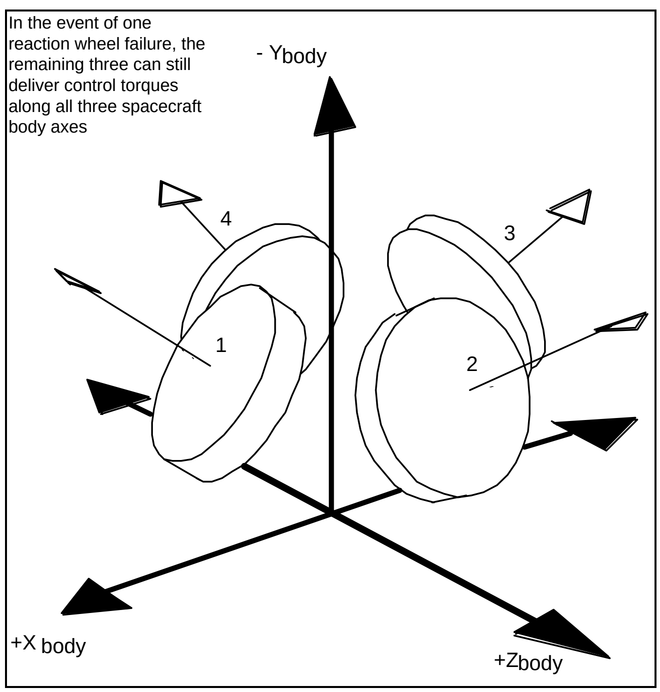


Figure 4.2-8 RWA Orientation Along the Spacecraft Body

<u>RWAs Only Accept Control Signals from the In-Control ACE</u>
The RWAs accept commands from one ACE at a time, which is designated as the "In Control" ACE. Each ACE provides a bilevel signal to each of the RWAs that indicates whether that ACE

thinks it is In-Control. The RWA electronics performs an exclusive-or (XOR) of the two inputs, and if the result is zero, the RWAs accept commands from ACE A. If the XOR result is one, the RWAs accept commands from ACE B. The point of this XOR is to assure that the RWA accept commands from only one ACE at a time, and at the same time still account for the possibility that a given ACE may have failed and have incorrectly set its bi-level signal.

Nominally, ACE-A is "In-Control" while ACE-B is monitoring ACE-A for state of health. The signals coming from the ACEs would make it so that the reaction wheels would take commands from ACE-A. If ACE-A has problems, it will suspend delivery of the "I'm OK" signal to ACE-B. ACE-B will respond to this situation by altering it's own bi-level. As a result, the reaction wheels would subsequently take commands only from ACE-B.

ACE-A bi-level	ACE-B bi-level	XOR	ACE That RWA will accept control commands from
0	0	0	A
0	1	1	B
1	0	1	B
1	1	0	A

Table 4.2-2 RWA-ACE In Control

4.2.1.9 Magnetic Torquer Bars

The TRMM spacecraft uses three MTBs to perform momentum unloading of system momentum. The MTBs produce a magnetic dipole moment which reacts with the Earth's magnetic field resulting in a torque on the spacecraft. Each MTB has two separate windings with either providing sufficient magnetic dipole strength to meet system momentum unloading requirements. The dipole moment of a single winding is 300 amp-meter squared, with a double winding of 360-370 amp-meter squared.

Each ACE drives one winding per bar. The windings are hard-wired to a specific ACE and are not cross-strapped, as shown in Figure 4.2-9. However, a drive signal to each winding on a bar can come from each ACE simultaneously.

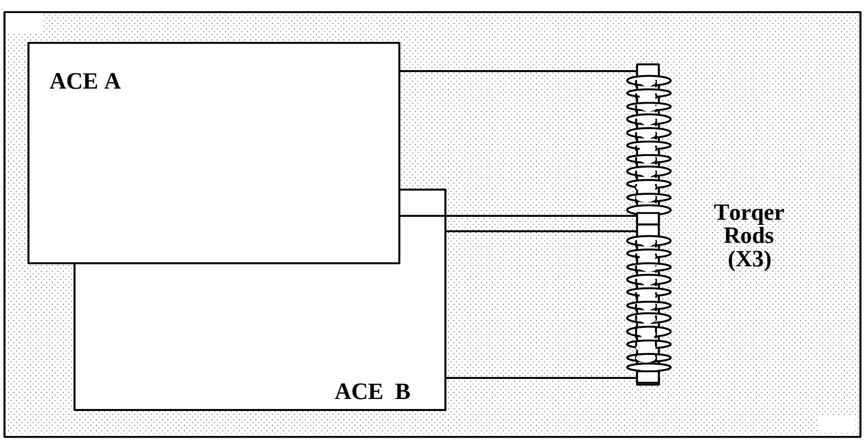


Figure 4.2-9 ACE/MTB Interface Diagram

4.2.1.10 Engine/Valve Driver

The EVD is the interface between the ACE and the RCS thrusters. The EVD is a single, fully redundant unit cross-strapped to both the ACE and to the Rocket Engine Modules (REMs). The EVD is capable of firing thrusters individually or in groups, continuously or one-shot firings, for pulses as short as 125 ms (8 Hz). The EVD is slaved to ACE commands and provides EVD and RCS status telemetry to the ACE. The EVD contains two Isolation boards, two Logic boards, two Thruster boards, and assorted other cards.

<u>Thruster Driver Boards Only On During Maneuvers</u>
During normal operations both Logic boards and one Isolation card will be powered ON continuously. Thruster drivers will not be powered ON until just prior to scheduled Delta-V maneuvers. Power interfaces are through the GSACE D-bus Control Module, with separate interfaces to each EVD from both GSACE A and B. Figure 4.2-10 shows EVD interfaces.

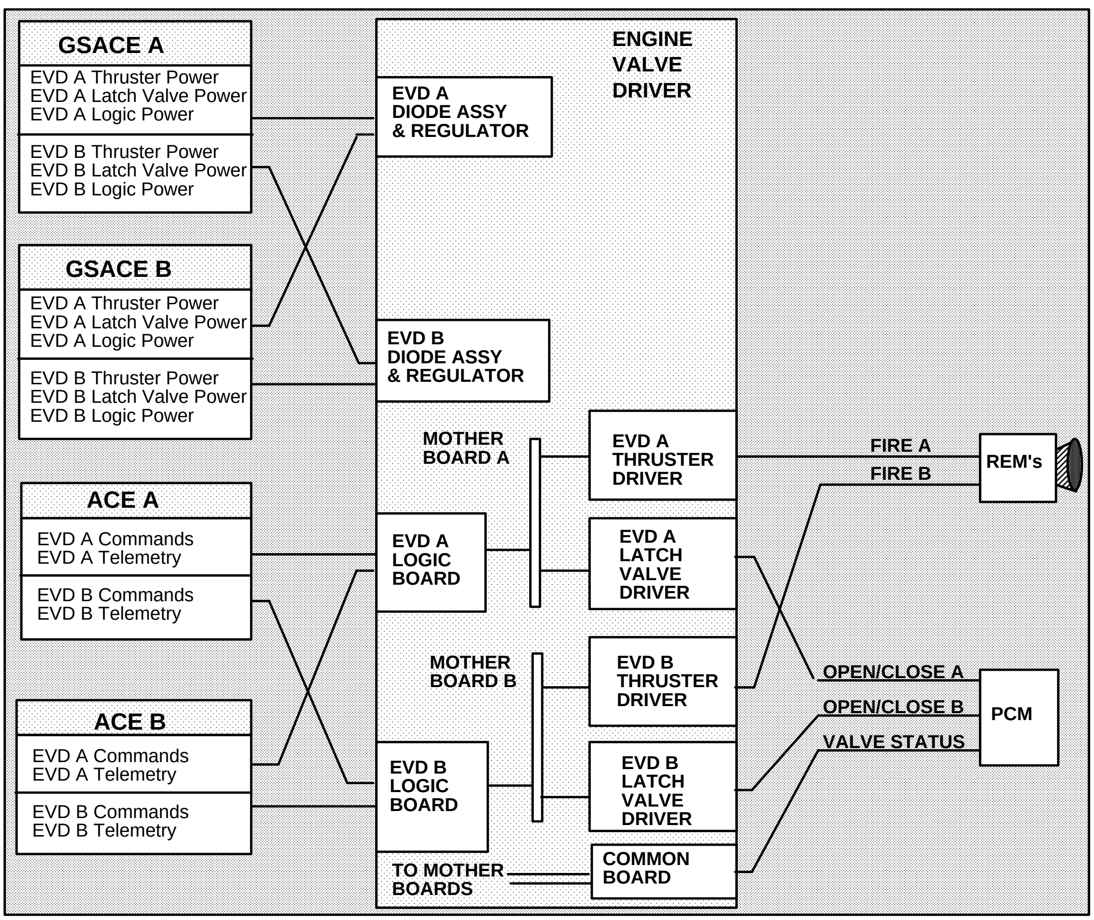


Figure 4.2-10 EVD Interfaces

ACS/EVD Interaction During a Burn

There is a protocol defined between the ACS and the EVD to assure veracity of thruster commands generated by the ACS. To commence operations, the ACS enables only the select subset of thrusters that will be needed for the maneuver. The selection is based on commands stored onboard by the FOT during set up for the burn as well as the spacecraft orientation. This enable signal is sent only once, it remains valid throughout the burn.

Once it is in Thruster Mode, the ACS will send one command packet to the ACE every .125 seconds (8 Hz). This will continue for the duration of the burn. Each command consists of a command word that specifies which thrusters to fire, as well as the complement of this word.

If the command word is verified, the ACE issues the command to the EVD, which sets up the appropriate thrusters to fire. The EVD echoes this command back to the ACE. If the echoed command agrees with the transmitted command, the ACE issues a fire command to the EVD. If the echoed command does not agree, the Thruster mode is aborted. The EVD will automatically stop firing after 200 ms if it has not received another burn command after 125 ms. After completion of a thruster operation the EVD Thruster Driver board will be powered OFF by the FOT via stored command. The EVD Control Diagram in figure 4.2-11 shows command flow between ACS, ACE and EVD during thruster operations.

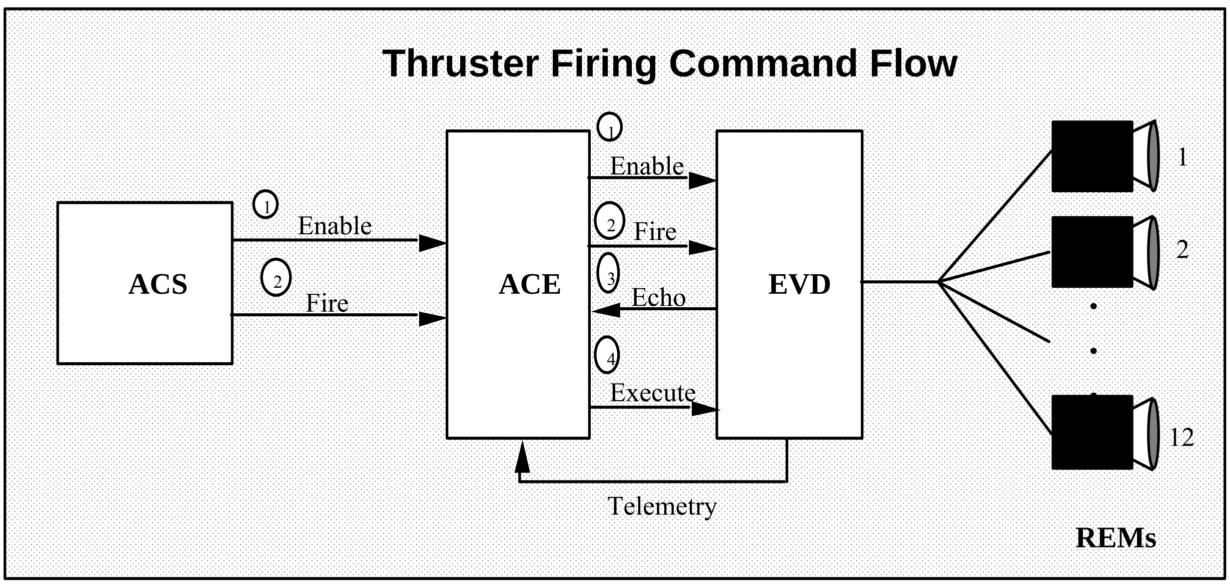


Figure 4.2-11 EVD Control Diagram

4.2.2 ACS Control Modes

The TRMM spacecraft has several control modes supported by the ACS and ACE processors . The Acquisition, Mission, Calibration, and Thruster modes are controlled from the ACS processor. The ACE SafeHold mode is controlled from the ACE.

Detailed descriptions of each control mode is provided in the ACS Software User's Guide [TRMM-712-184]. The present discussion will focus on the background information that was implicit in the User's Guide, as well as on aspects that have direct bearing on operations. For a full discussion of each mode, such as the sensor complement used to maintain the mode, or the detailed rules for autonomous transition between modes, see the User's Guide.

This section will introduce the major control modes. Discussion of the submodes within each of these modes is provided in the next section.

Mission Modes
These will be used most during the mission. There are two: Nominal Mission Mode and Yaw Maneuver Mode. Both of these modes keep the spacecraft nadir pointing. Nominal Mission Mode maintains a fixed yaw angle, while the Yaw maneuver mode allows the spacecraft to transition between fixed yaw angles. While in this mode TRMM will be at a fixed spacecraft yaw around the nadir vector.

Acquisition Modes
These are intended to be a series of modes that can be used in sequence to get the observatory into Mission Mode. The Acquisition modes are: Sun Acquisition, Earth Acquisition, and Yaw Acquisition. These modes work together as a string. The ACS control algorithms are designed to allow for a smooth attitude transition between these modes. After the last of these modes, when the right conditions are met, attitude control will be automatically handed off to Mission Mode.

For example, at separation from the launch vehicle, there are no guarantees about the orientation of the observatory. By going through this sequence of modes, the spacecraft will go from pointing in no particular direction at all to pointing with the +Z axis pointing down toward the Earth and our choice of +X or -X pointing along the direction of flight.

Calibration Mode
The Calibration mode supports CERES Deep Space Calibration operations. It maintains the attitude in an Inertial-Hold.

Thruster Modes
Use of the RCS is limited to the two thruster modes. They are the Delta-V and Delta-H Modes. The Delta-V Mode will be used regularly to maintain the orbit of the spacecraft. The Delta-H mode is an attitude control mode that may be used in a number of backup situations.

Standby Mode
The ACS processor also supports a non-control mode, Standby mode, used when the ACE takes over in SafeHold. SafeHold mode is implemented independent of the ACS processor. The following paragraphs discuss the function and transitions of all control modes resident in the ACS. Section 4.2.4.5 discusses ACE SafeHold.

Mode Transitions

In flight, we will not be freely jumping from one attitude control mode to another. Some transitions will be controlled solely by ground command, while other controlled by flight software, some transition are not allowed at all. For example, after separation, the spacecraft will automatically transition from Standby to Sun Acquisition mode, but the transition from Sun Acquisition to Earth Acquisition will be done via ground command. On the other hand, the transitions from Earth Acquisition to Yaw Acquistion to Mission mode will occur automatically, as the attitude and rates come within acceptable limit conditions for the transition. As a final example, once we are in Mission Mode, when it comes time to do Delta-V maneuvers, a command will be sent to the spacecraft that includes the maneuver duration. The ACS will then transition to this mode and then exit to Earth Acquisition Mode once the burn duration is complete.

Another way the spacecraft transition may take place is if the Failure Detection and Correction (FDC) routine within the ACS task detects a problem. For example, if the spacecraft is in Nominal Mission Mode, but the Pitch, Roll or Yaw attitude or rates are out of limits, it will put the ACS into Sun Acquisition Mode.

A third way that transition between modes can be control is via the TSM task residing on the SC processor. It is described in section 4.1.4.10. Were the TSM to detect that the SC Processor is not performing nominally, it would disable the ACS absolute time processor. This is where commands for future transitions into Delta-V mode would be stored by the ground. Disabling it would cancel any upcoming Delta-V maneuvers.

Finally, a cold start will force the ACS to start over in standby mode, while a warm start will only have an effect if the ACS is in thruster mode. In the event of a warm start, the ACS will transition out of thruster mode into an acquisition mode.

In summary, there are five possible reasons why a mode transition may occur. They are i) <u>Autonomous</u> transition by the ACS ii) <u>FDC</u> requested transition due to anomaly iii) <u>TSM</u> disable of transition iv) <u>FOT</u> commanding a transition and v) <u>Cold/Warm</u> Start. Figure 4.2-12 summarizes the control modes, their submodes and the reasons for transition between them.

4.2.2.1 Standby Mode

ACS Standby mode provides a non-control ACS mode entered autonomously when the ACE takes over in SafeHold. With the exception of launch, one can say that if the ACS is in Standby mode, then the ACE is in SafeHold control mode. In Standby mode, the ACS provides no attitude control. All spacecraft and Solar Array pointing is accomplished through SafeHold interfaces and ACE control algorithms. Although no ACS controller is present, spacecraft attitude is monitored and coarse HGA pointing is possible.

Standby mode may be entered by an FDS cold start on the ACS processor, by ground command, or by Failure Detection and Correction (FDC) request. Standby mode may be exited by either ground command or by detection of separation during launch operations. To exit from Standby mode, the ACE "I'm OK" signal must be enabled, the spacecraft separation counter must equal 0, the ACEs must not be in SafeHold, and at least three reaction wheels must register non-zero voltages.

4.2.2.2 Sun Acquisition Mode

Sun Acquisition mode provides a power and thermal safe orientation for the spacecraft in which the ACS processor executes equations similar to the ACE SafeHold control algorithms. There is, therefore, no change in attitude in transitioning from ACE to ACS control in SafeHold recovery. The solar arrays are commanded to the index position. During the sunlit portions of the orbit, CSS data are used for position errors and IRU data are used for rate errors. During eclipse, IRU data are used for both position and rate information. As in Standby mode, coarse HGA pointing is possible. Sun Acquisition mode uses the full capability of both MTBs to improve momentum unloading capabilities.

All control modes transition to Sun Acquisition mode in the event of an ephemeris failure. Sun Acquisition may be exited only to Earth Acquisition mode (via ground command only), and valid spacecraft ephemeris and ESA stabilization is required prior to attempting the transition to Earth Acquisition.

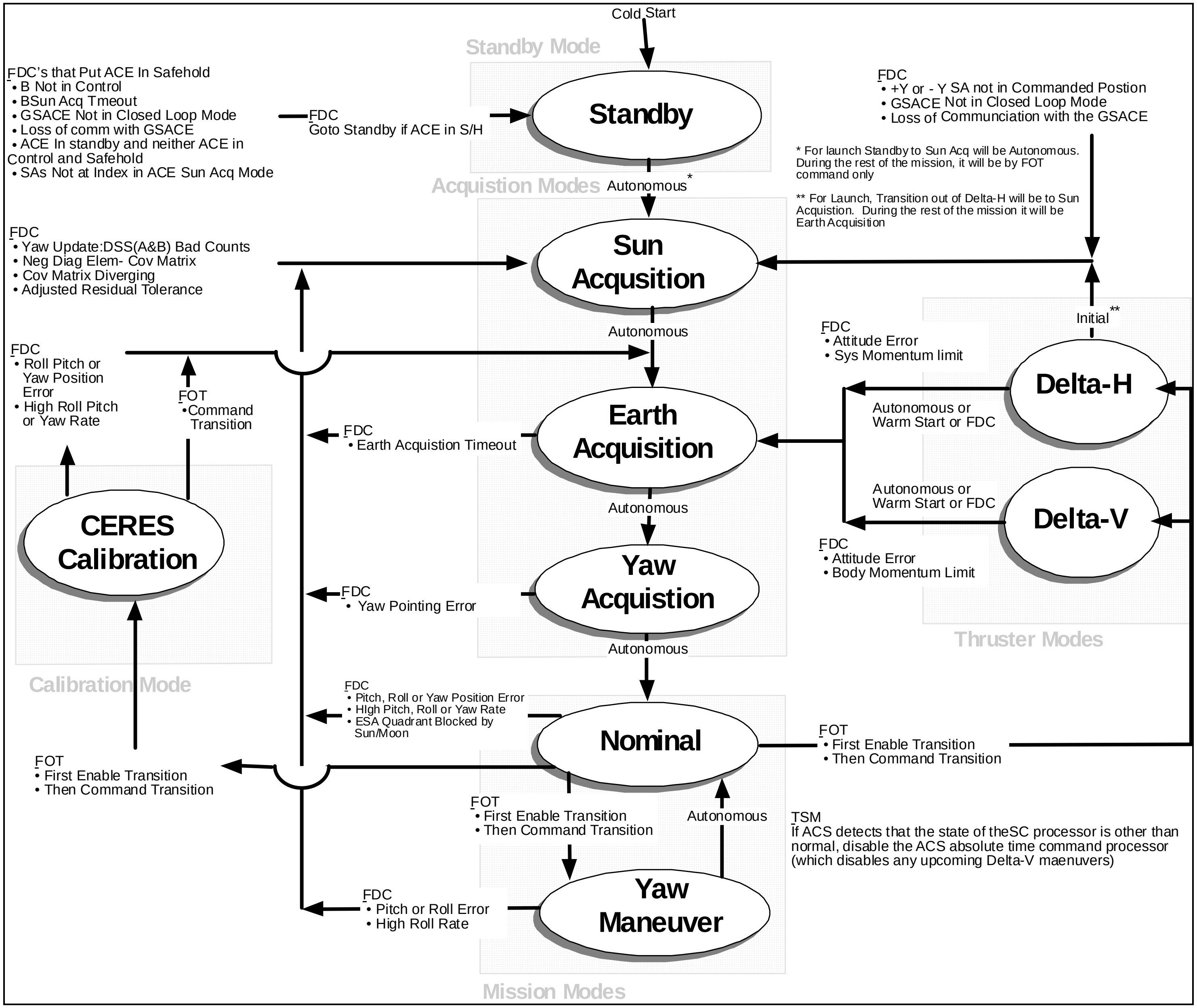


Figure 4.2-12 ACS Control Mode Transitions

4.2.2.3 Earth Acquisition Mode

Earth Acquisition allows for the spacecraft to point the spacecraft +Z axis towards the Earth, or to acquire the nadir, from any orientation. Earth Acquisition mode is entered from Sun Acquisition mode by ground command or autonomously at the end of Delta-V, Delta-H, or CERES Calibration modes. Valid spacecraft ephemeris is required to enter into Earth Acquisition mode when entering from the Sun Acquisition mode, since all following control modes require valid ephemeris for SA
pointing. Coarse SA pointing is possible when the Sun is available. If entering from Sun Acquisition, the SAs are feathered. Otherwise, the SAs are left alone upon entry from other control modes. When entering Earth Acquisition mode, a timer is started and acquisition of the Earth must occur within 60 minutes (as specified in ACS FDC tables). Otherwise, the ACS will transition back to Sun Acquisition mode.

In Earth Acquisition Mode, if there is Earth presence, the ACS control algorithms will use the nominal Earth Acquisition control law. If none of the four ESA detectors are on the Earth, a roll rotation is initiated, which will drive the +Z-axis toward nadir. Earth Acquisition is exited autonomously to Yaw Acquisition mode when the roll and pitch errors are both less than 1° for 340 continuous control cycles, and the ESA is in Fine data processing mode.

4.2.2.4 Yaw Acquisition Mode

Yaw Acquisition is entered autonomously from Earth Acquisition and always occurs as one step in the transition from Delta-V to Mission mode. The Yaw Acquisition control mode aligns either the +X or -X spacecraft axis with the velocity vector, depending on target orientation. SA pointing is performed when in orbit day, and SAs assume nominal pointing once the sensed yaw error is less than a specified value. As with Earth Acquisition, a timer is started when entering Yaw Acquisition mode and acquisition must occur within 60 minutes. Otherwise, the ACS will transition to Sun Acquisition mode. The Yaw Acquisition mode is exited autonomously when the yaw error is less than 2.5° for 10 continuous control cycles.

4.2.2.5 Mission Mode

Mission, or Normal, mode provides the primary science data collection control mode. In Mission mode, roll and pitch errors are computed from ESA data, and integrated yaw gyro rates provide the yaw attitude error. Yaw error updates are made (nominally) twice per orbit by the DSSs. The MTB current is limited to minimize magnetic contamination of the science instruments. Fine HGA pointing is available, and the SAs are driven to the Sun during orbit day and feathered during orbit night.

Three possible attitudes can be commanded in the Mission mode: fore (+X forward), aft (-X forward), and yaw (-Y forward). These modes are provided to maintain the VIRS instrument in a thermally safe environment and to allow for PR antenna pattern measurements. In transitioning among the three orientations, the direction of rotation is determined by the ACS (not by ground command) and transition is accomplished through the Yaw Maneuver sub-mode. Direction of rotation is always to keep the +Y side away from the velocity vector, to avoid contamination of the VIRS instrument.

The ACS maintains three axis stabilization controlled to the horizon bisector frame. Each control axis employs a position-integral-derivative control law sampled at 2 Hz. A momentum distribution law is utilized to minimize the differences in wheel speeds. Reaction wheel torque commands are modified by the resultant of the distribution law, which does not introduce momentum to the spacecraft since the vector sum of the torque distribution commands is equal to zero.

4.2.2.6 Yaw Maneuver Sub-Mode

The Yaw Maneuver sub-mode provides a means to change attitude orientation in Mission mode to either +X, -X, or -Y orientations. Yaw Maneuver is entered directly from Mission mode and, if successful, autonomously returns to Mission Mode upon maneuver completion. ACS control is similar to that in Mission mode where roll and pitch are controlled through the ESA and yaw is based on propagated IRU data. If a Yaw maneuver occurs at orbit dawn or dusk when a DSS update should occur, the DSS update is disabled and the previous value is maintained with no affect to ACS operations. Fine HGA pointing and SA pointing functions available in mission mode are also available in the Yaw Maneuver sub-mode.

Section 7.1.1.5 describes the planning and scheduling operations of Yaw maneuvers.

4.2.2.7 CERES Calibration Mode

CERES Calibration mode provides an inertially fixed mode used to point the CERES instrument to deep space. Attitude errors are developed by integrating the gyro rates. The CERES Calibration mode is entered by ground command from the Mission mode and must be entered at orbit noon. There is no ACS check on the entry of CERES Calibration mode and it is the responsibility of the FOT to ensure entry is made at orbit noon. The SAs remain fixed while in the CERES Calibration mode, however HGA pointing is possible based on the frozen, commanded attitude and sensed pointing errors. The CERES Calibration mode must also be exited by ground command.

To accomplish a CERES Deep Space Calibration, ATS commands originating from the S/C processor will be used. The FOT will ensure that the ACS transitions to the inertial hold mode at orbit noon (calculated as midpoint between sunlight entry and exit), and will transition back to Earth Acquisition mode one orbit later. During a CERES Deep Space calibration, the FDF will also perform additional ESA calibrations to ascertain the current state of ESA fogging.

4.2.2.8 Thruster Mode

Thruster mode is the means by which the ACS employs the RCS thrusters. This mode is intended to be used for two purposes. The main purpose is to employ a complement of thrusters to regularly raise the TRMM orbit when it is about to go below mission requirements. The second use of the RCS is as an alternative means to control the attitude of the spacecraft.

While in thruster mode, the RCS is the sole means used for attitude and orbit control of the observatory. Spacecraft attitude sensed by the IRU, and the desired attitude is maintained by thrusters . Reaction wheel control torques are set to zero. Also, the ACS, operates at 8 Hz, generates thruster firing commands to the RCS via the ACE/EVD interface.

ACS Activities Prior to Maneuvers

It is planned to power off the EVD Thruster Driver board between maneuvers. Therefore, prior to the maneuver, one must power these boards on. This is described in detail in the EVD section of this document. Also, in addition to commanding the transition to Thruster mode, one must first enable the transition. This two step process is to ensure against accidental commanding.

Thruster Selection Table

Not all of the thrusters will be active during a given maneuver. Only a select subset are needed in most cases. During all maneuvers, the roll thrusters will be required. They are used by the ACS software in conjunction with IRU information to maintain the roll angle of the spacecraft. Among the rest of the thrusters, the actual selection needed depends on which thruster sub-mode is being used, the spacecraft orientation and the mission phase. Details are discussed in each of the sections below that discuss the thruster sub-modes. This table may be modified on-orbit, in the event of a thruster failure.

Thruster Firing During the Burn

Once the ACS is in Thruster Mode, commands will be sent every 0.125 seconds (8 Hz) to the ACE, which in turn will direct the EVD to open the RCS thrusters. If for any reason, this flow of commands is interrupted, the ACE will shut down operations of the thrusters.

ACS Control Algorithms

A control algorithm is encoded within the ACS software to intermittently turn on the Roll thrusters during thruster maneuvers to control spacecraft attitude roll errors. The process is called 'On-Modulation'. Upon execution of a burn command the first burn pulse will occur after 125 ms. The last burn pulse will be terminated after 125 ms by the ACS sending a Thruster_OFF command (rather than continuing for 200 ms).

During Delta-V burns, the same thrusters being used to adjust orbit will be intermittently turned off via an ACS control algorithm that is intended to control pitch and yaw during the maneuver. This process is called 'Off-Modulation'. If we command the thrusters to burn for, say, 30 seconds, that means that we are asking for 30*8 = 240 counts of firing time. You may not get 240 counts of thruster on time however. After 30 seconds, the thrusters will be turned off. During that 30 seconds, if select thrusters are off modulated, then they won't reach 240 counts. Instead you must consult telemetry. There are counters for each thruster that tell you the number of burn pulses that were actually commanded by the ACS.

While in Thruster mode, the ACS will continue to propagate TRMM and TDRS ephemerides and provide HGA pointing capability to any one of the four TDRS satellites. Propagation of the COMETS ephemeris is not possible and will require reloading after a Delta-V and prior to the next COMETS event. HGA commanding continues at a rate of 2 Hz, and not 8 Hz as with other ACS control functions. No SA pointing is performed while in Thruster mode.

4.2.2.8.1 Delta-V Sub-Mode

Delta-V is a thruster sub-mode which provides an orbit adjust capability to maintain the spacecraft to within the altitude requirements for science collection, perform an initial orbit adjust maneuver after launch (380 km to 360 km), and to perform the EOL ocean disposal.

Use of ATS and RTS

The Delta-V mode will be used regularly. Maneuver Commands will be loaded onboard, using a combination of the SC Processor and ACS Stored Command Processors. Loaded into these processors will be commands to power on the EVD, turn on cat beds and command entry into the Delta-V Mode.

Entry in to Delta-V Mode
To command the ACS into Delta-V mode, one must also specify the expected position and velocity of the vehicle at the end of the maneuver. In addition, one must also specify the Drag Scale Factor, the Exospheric Temperature as well as the Geomagnetic Activity Index. This information is provided to the FOT by FDF prior to each maneuver. Finally, the maneuver duration, expressed as the number of 0.125 second (8 Hz) pulses required, must be specified.

In addition to the required information needed to command transition to Delta-V, the spacecraft must also be in a valid spacecraft orientation. Either the +X or the -X axis of the spacecraft must be forward. Otherwise, the ACS will determine that there is no valid selection of thrusters with which to execute the maneuver. This is discussed more in the next few paragraphs.

Thruster Selection
In addition to the roll thrusters, the ACS will select whether the LBS or the ISP thrusters will be used. The choice that the ACS will make depends on which phase of the mission that the ACS has been commanded to observe as well as which axis of the spacecraft is pointing forward along the velocity vector of the orbit.

During the Beginning of Life, TRMM will have been placed by the launch vehicle into an orbit that is higher than the required mission orbit. To get into mission orbit, the thrusters pointing away from the direction of flight must be fired to decrease the orbit altitude. Once the mission orbit has been reached, the Mission Phase begins. In this phase, regular reboosts will be done to counteract drag. In this case, the thrusters pointing toward the direction of flight must be fired to increase the altitude of the orbit. Finally, when it comes time to dispose of TRMM in the ocean, the End of Life Phase of the Mission takes place. During this time, a series of burns directed away from the direction of flight will be done to drop the altitude.

Based on this profile, the following table is programmed onboard. The (ground commanded) mission phase and the current spacecraft orientation is used to determine which bank of thrusters may be used. Note that if neither the +X or -X are forward, there are no valid thrusters. In this case, the ACS will ignore commands to transition into Delta-V mode.

Spacecraft Orientation	Mission Phase	Beginning of Life Phase	End of Life Phase
+X Forward	LBS	ISP	ISP
-X Forward	ISP	LBS	LBS
-Y Forward	None	None	None
Inertial	None	None	None
Unknown	None	None	None

Table 4.2-3 Delta-V Thruster Selection

Exit From Delta-V Mode
The control mode that the ACS will transition to after completion of Delta-V mode is specified in ACS table 90. This will be set for Earth Acquisition Mode. The ACS will automatically go into Earth Acquisition Mode once the duration that the FOT specified for Delta-V Mode has expired. At completion of the burn, the ACS will take up and commence propagation of the EPV that was uplinked as part of the command to go into thruster mode. The ACS will meet mission pointing requirements within 7 minutes after each maneuver burn.

Example Maneuver
An example of operations is shown in table 4.2-4. This example assumes that the specified burn duration was one second.

<u>Use of the Post Maneuver EPV</u>
Included as part of each Delta-V command will be the predicted post-burn TRMM ephemeris in Extended Precision Vector (EPV) format. The epoch of the EPV will be scheduled at the completion of the maneuver. For a two-burn maneuver sequence, there will be two EPVs executed. In the event of a failure of the orbit adjust maneuver, the ACS performs a linear interpolation between the pre-maneuver and post-maneuver ephemerides based on the fraction of the burn completed, to create a best-estimate ephemeris to allow continuation of spacecraft operations. However, a new ephemeris will be uplinked as soon as tracking data is available (approximately 8 hours later). The new EPV will be based on definitive tracking data after the maneuver has been completed.

Delta-V planning and scheduling operations are described in section 7.1.1.4.

Relative Time	Activity
00:00:00.000	Delta-V Command executes from ACS ATS, 8 pulses are specified. ACS transitions to 8 Hz processing (BCRT Interrupt) ACS enters Delta-V mode (BCRT Interrupt)
00:00:00.125	1st burn command executes
00:00:00.250	2nd burn command executes
00:00:00.375	3rd burn command executes
00:00:00.500	4th burn command executes
00:00:00.625	5th burn command executes
00:00:00.750	6th burn command executes
00:00:00.875	7th burn command executes
00:00:01.000	8th burn command executes
00:00:01.125	ACS terminates burn ACS executes TRMM EPV update ACS transitions to 2 Hz processing
00:00:01.500	ACS enters Earth Acquisition mode

Table 4.2-4 One Second Delta-V Breakdown

4.2.2.8.2 Delta-H Sub-Mode

Delta-H is a thruster sub-mode which uses a combination of fore, aft and roll thrusters to produce pure torque without orbit changes. The Delta-H control mode may be used for excessive tip-off rate unloading, EOL momentum management, backup reaction wheel unloading, or as a backup yaw maneuver control mode.

<u>Entry into Delta-H Sub-mode</u>
To be able to command the ACS into Delta H Sub-mode, the FOT must specify which torque control algorithm: Momentum Unload, Yaw Turn or Reaction Wheel Unload that the ACS should use while in Delta-H Mode. Furthermore, if the choice is Yaw Turn, the target spacecraft attitude: +X Forward, -X Forward or -Y Forward, must also be specified. Finally, the duration

that the spacecraft is allowed to remain in Delta-H mode must also be specified. Rather than seconds, it is expressed as the number of 0.125 second (8 Hz) pulses required. The maximum duration is 8191 seconds.

<u>Thruster Selection</u>
For most phases of the mission, all of the thrusters will be used to maintain momentum control. In the End of Life Phase, the choice of whether to use the ISP or LBS thrusters depends on the orientation of the spacecraft. Table 4.2-5 shows the shows the selection rules employed onboard.

<u>Exit From the Delta-H Sub-mode</u>
The control mode that the ACS will transition to after completion of Delta-H mode is specified in ACS table 90. In early mission, the FOT should set this to Sun Acquire Mode. Once into the Mission Phase, this should be set to Earth Acquisition. The ACS will automatically go into this mode once the duration that the FOT specified for Delta-H Mode has expired. Before this expiration, the transition can occur if the excess system momentum has been reduced sufficiently.

Spacecraft Orientation	Mission Phase	Beginning of Life Phase	End of Life Phase
+X Forward	All	All	ISP
-X Forward	All	All	LBS
-Y Forward	All	All	None
Inertial	None	None	None
Unknown	All	All	None

Table 4.2-5 Delta-H Thruster Selection

<u>When the Delta-H Mode is Needed</u>
Ordinarily, it is not expected that we will need to use Delta-H mode. Tip off rates should be low enough that the reaction wheels can absorb them. Momentum management and reaction wheel unloading are done by the magnetic torquer bars and yaw maneuvers can be done by the Yaw Maneuver Sub-Mode.

<u>One Shot Firing</u>
The ACS also provides a one-shot thruster firing capability to be used in the event of high tip-off rates after deployment from the H-II. The ground will select and fire one or more thrusters to take out some component of system momentum. All inputs and direction for one-shot thruster firing will come from Code 712.

4.2.3 ACS Operation

This section will discuss key aspects of how the ACS operates and how the FOT is expected to interact with the ACS.

4.2.3.1 Initialization

The ACS task is started by the FDS, on the ACS processor, when the ACS processor is commanded to Normal mode. The ACS will transition to Standby mode upon power ON or after any cold start. If a warm start occurs, the ACS remains in its current mode unless the current mode is a Thruster mode (Delta-V or Delta-H). If a warm start occurs while the ACS is in Thruster mode, the ACS will transition to Earth Acquisition mode.

4.2.3.2 Command and Telemetry Processing

The ACS validates the command packet header information and rejects improperly formatted commands. Upon successful command validation, the secondary header and application data portion of the CCSDS command packet are copied to the ACS internal buffering system, which are executed once per control cycle. The internal buffer is cleared after command execution by setting all bytes to zero. The FDS buffers up to 20 incoming commands for the ACS in the ACS command pipe queue.

Note that in the Thruster mode (8 Hz control cycle) up to 8 commands may be processed per second. Operationally, the FOT will ensure no additional real-time or stored commands will be executed by the ACS or ACE while in the 8 Hz control cycle (Thruster mode) other than those directly relating to the orbit maneuver.

The ACS telemetry generator outputs to the FDS all telemetry for the current control cycle. Telemetry processing occurs at the end of each control cycle in order to downlink all data generated for the cycle, and to service any requests for ACS packet dumps or FDS housekeeping requests received by the command processor. The ACS provides both 8 Hz and 2 Hz telemetry packets during Thruster mode. This is allowed by skewing the timing of the 2 Hz and 8 Hz interrupts by 62.5 ms (one-half of an 8 Hz cycle).

The ACS also generates asynchronous telemetry packets. These packets include ACS and ACE spacecraft event messages, which are either downlinked in real-time or passed to the FDS Data Storage task and recorded in bulk memory. The ACS provides up to 25 spacecraft event messages per control cycle. Each status message consists of a 2 byte message code with a four byte parameter. The FDS provides the capability to filter event messages, which may be controlled by ground commands. There also exist packets generated only upon command, which provide ACS configuration, statistics, and diagnostics information. ACS configuration, statistics, and diagnostics packets will be downlinked during each real-time contact.

All ACE telemetry is processed on the ACS processor. No data will be downlinked or recorded from the ACE or ACS sensors and actuators when the ACS processor goes down.

4.2.3.3 Failure Detection and Correction

ACS FDC provides an internal means for ACS anomaly determination and resolution. There are two components of the FDC. First is failure detection. Various points within the ACS task check for anomalous conditions that apply to that point in the process. When problems are identified, a flag is passed to the correction component of the FDC.

Occurring near the end of the ACS task, the correction component collects together all of the detection flags thrown during task execution. The correction process tabulates the number of consecutive times the given error has been detected. Thresholds have been set for each kind of error. Once the threshold is passed for a given error, the corresponding corrective action is executed.

Some errors have more than one threshold, after just a few consecutive errors have been detected, one action may be initiated. If, however, the problem persists and a second threshold is reached, another, more drastic action is commanded. Up to three levels of detection and correction are supported. Actions include operations such as switching the choice of IRU from primary to redundant, or placing the ACS in Sun Acquisition mode.

As a general rule, the FDC attempts appropriate action to maintain Mission mode and science data collection, but will command the spacecraft to a safe mode of operation as required. The FDC is single fault tolerant, and does not attempt to detect and correct multiple failures occurring simultaneously. If an anomaly exists and a hardware problem is not identified it is assumed that a software problem exists.

FDC software is table driven, with limit counters and specifiable actions. Individual FDC tests and actions can be enabled or disabled, or FDC may be globally enabled or disabled.

4.2.3.4 Ephemeris Propagation

The ACS maintains an ephemeris propagation capability in support of HGA and SA pointing. TRMM, four TDRSs, and COMETS positions and velocities are propagated using a fourth order Runge-Kutta propagation routine. Individual TDRS and COMETS ephemeris propagation may be separately enabled and disabled (via table load). The spacecraft propagation is accurate to within 1 km radial, 40 km down range, and 1 km cross range. Real-time operations regarding EPV loads are discussed in section 6.3.5, and off-line operations are described in section 7.2.3.

Continuity checks are performed by the ACS on updates to TRMM, TDRSS, and COMETS ephemerides. Time, position, and velocity must be within specified limits or the spacecraft will reject the update and continue propagation of the old ephemeris. In the event of a failure in ephemeris propagation, a new ephem must be jam loaded. After the new ephemeris has been uplinked and accepted by the ACS software, the limits will be returned to normal values. The propagation of TDRS and COMETS ephemerides differs from TRMM propagation in that spacecraft drag calculations for TDRS and COMETS are disabled.

Propagation of solar and lunar ephemerides are also performed based on a low precision formula for geocentric coordinates of the Sun and Moon. These data are used for eclipse logic, density model propagation (Sun hour angle and declination), ESA logic, and SA control. The ACS maintains a software eclipse flag which is compared to data received from ACS hardware. No updates to solar and lunar ephemerides are required during the mission (including at millennia cross-over).

4.2.3.5 Deployables Management

In addition to attitude determination and control functions, the ACS also performs management of the SAs and HGA. The ACS processes SA and HGA sensor data via the GSACE and generates gimbal position and velocity command data sent to the GSACE for transmission to the SAs and HGA. Validation of the commanded SA and HGA positions is performed while tracking or stopped, but is not performed while slewing.

4.2.3.5.1 Deployables Control Modes

In ordinary operations, the ground team will not directly command the HGA gimbals nor the Solar Array. Instead, commanding will be done by the ACS. There are four modes that the ACS uses for commanding the GSACE. These are Track, Feather, Stop and Index. The first three apply to the HGA. All four apply to the Solar Arrays.

<u>Track</u>
When the ACS is using the track mode, the deployable will track it's target. For the HGA, the ACS will use its onboard propagation of the TDRS or COMETS position. From this it will compute the necessary commands to send to the GSACE that will point the HGA towards TDRS. For the SA, the ACS will compute the angle that will collect the most sunlight and rotate the arrays to that angle These SA positioning commands are then passed to the GSACE.

<u>Feather</u>
When the ACS is using this mode to control a given deployable, the deployable will be oriented edge on to the direction of flight. As TRMM plows through the atmosphere, this position will minimize the surface area presented to the wind of particles blowing past the spacecraft. For the HGA, this means the α and β angles are zero. For the SA, this is also is zero for both the -Y and +Y motors.

<u>Stop</u>
As the name implies, this command stops the deployable dead in its tracks. It can be commanded for either the HGA or SA. It will not be used in ordinary operations.

<u>Index</u>
This command only applies to the Solar Arrays. This command will tell the arrays to move from wherever they are pointing right now and go to the 'Index' position of the solar arrays. This particular command is used when the ACS is in Sun Acquisition Mode or when the ACE is in Safehold mode.

4.2.3.5.2 Solar Array Control

The selection of solar array pointing mode is largely controlled by ACS algorithms. No intervention is required by the ground. Table 4.2-6 summarizes the SA control and configuration for each ACS control mode.

Control Mode	SA Configuration
ACE SafeHold	Indexed
Standby	Indexed
Sun Acq	Indexed
Earth Acq	Feathered if entered from Sun Acq. Left alone if entered from other modes
Yaw Acq	Nominal pointing once yaw errors less than tolerances.
Mission	Nominal pointing
Yaw Maneuver	Nominal pointing
CERES	Stopped
Delta-V	Stopped
Delta-H	Stopped

Table 4.2-6 SA Configuration to Control Mode

Automatically Switching Between Track and Feather

The two SAs are controlled by the ACS control algorithms. Each SA is individually commanded. When the SA is in sunlight, the ACS will employ Track Mode. When TRMM passes into earth shadow, the ACS will automatically Feather the SA. Just prior to re-entry into sunlight, the ACS will automatically slew the SA to resume Track of the Sun. Pointing and control of the SAs by the ACS is performed using propagated solar and TRMM ephemeris data.

Index Position is Used During Safehold
Index places the solar arrays in a fixed position with respect to the spacecraft. Subsequently, pointing control of the entire spacecraft is done by the ACE via the CSS's. The normal to the SAs is maintained to 10° +/- 10° of the Sun line during tracking to maximize power. During slewing to and from the feathered or indexed positions the slew rate is damped to less than 2° per second to avoid disturbance.

Ground Commands to the SA
SA ground commands include SA Stop and SA Index. These will not be needed ordinarily. SA ground commands remain in effect until another ground command is sent, the ACS issues another command in transitioning from one control mode to another, or orbit night is entered. One can however disable ACS control of the SA if needed.

SA Positions
To prohibit the movement of the SA beyond an allowable range, a number of safety measures have been implemented. Table 4.2-7 defines SA feathered and index positions, as well as hardware and software stop positions.

Solar Array Configuration	Gimbal Angle (degrees)
Feathered	0.0°
Indexed	-Y Solar Array = +90° +Y Solar Array = -90°
ACS Software Stop	±130.0°
GSACE Encoder Stop	±130.8° ±0.8°
GSACE Hard Stop	±174.0° ±1.0°

Table 4.2-7 Solar Array Positions

4.2.3.5.3. High Gain Antenna Control

The HGA is a two axis gimbal antenna system which tracks a TDRS or COMETS during real-time contacts.

Selection of Control Mode Is Done by the Ground
Unlike the SA, selection of the control mode of the HGA is commanded by the ground. For example, once the ground commands the ACS to use the Track mode for the HGA, it will continue to do so until further commands are sent from the ground. During non-contact times the ground will command the HGA into a feathered orientation to reduce aerodynamic drag. The HGA feathered position is zenith pointing.

Changing From Feather To Track
The maximum allowable slew rate for the gimbal motors is 90° per minute. Furthermore, to reduce disturbances to the rest of the spacecraft, the HGA slew rate is damped such that it will require four minutes to go from zenith to horizon (90°). Disturbances on the spacecraft must be less than 0.02° during slewing.

This plays an important role in contact scheduling. Between events, the antenna is pointing straight upward, with alpha and beta both equal to zero. When a real-time event begins, TDRS will appear on the horizon, which is a right angle to the Feathered position. This affects the commanding profile that the ground team uses for a TDRS contact. The command that the ground team places onboard to command the HGA from feather to track should execute prior to TDRS coming over the horizon. There must be enough time to allow the antenna to get into position to meet TDRS coming up over the horizon.

Slewing the HGA for Blind Acquisitions

During normal operations, the HGA is set to Track 4 minutes prior to the time TDRS has been scheduled to assure the HGA has time to get from zenith to the horizon. However, for blind acquisitions the MOC software will have a routine to allow the user to calculate the amount of time required to slew the HGA. This will be used as an indication that commands to start a real-time event, sent without telemetry verification, have been received and executed by the spacecraft.

HGA Usage In Various Attitude Control Modes

Prior to using the HGA to track a communications satellite, one must successfully uplink an EPV for that relay satellite, as well as one for TRMM. Even then, there are limitations on when the HGA will be useful. Not only does the target communications satellite have to be above the Earth's horizon, but it has to be within the range of motion of the HGA gimbals, and the relative rate of motion of the target has to be slow enough that the gimbal motors can keep up. In nominal mission mode, this is not a problem. TRMM is nadir pointing and the center of the gimbal range of motion is zenith. These together guarantee that if the target satellite is above the horizon, then it can be tracked. Other ACS Control modes do not carry such assurances. Table 4.2-8 describes HGA usage and limitations with respect to each ACS Control mode.

Control Mode	HGA Usage
Standby	If ACE in SafeHold, then that case applies. If not, there are no guarantees about HGA rates nor whether TDRS would be in view.
Sun Acq, ACE SafeHold	In Sun Acquistion Mode once it is complete, the roll axis direction is controlled, but rate about that axis is not. With TDRS above the Earth Horizon, it would arch over the gimbal range of motion again and again, moving at a rate that depended primarily on the roll rate of TRMM.
Earth Acq	For Earth Acquisition Mode, once it is complete, the spacecraft will be nadir pointing, but the yaw angle would be arbitrary. This means that it is possible that the target satellite is moving through the keyhole, where the HGA motors could not keep up.
Yaw Acq	In Yaw Acquisition Mode, the spacecraft should be nadir pointing. Once it is complete, the target satellite should not be in the keyhole, and so the gimbals should be able to track
Mission	Nominal pointing, target satellites can be tracked whenever they are above the Earth Horizon.

Yaw Maneuver	It is possible that the target satellite is moving through the keyhole, where the HGA motors could not keep up.
CERES	A target satellite that is in view at the start of the maneuver will still be in view one orbit later, unless it was just on the edge of view. During the night portion of the orbit however, the Earth may be in the way. Target satellites well above the Earth horizon during night will not be within the gimbal range of motion.
Delta-V	TRMM is nadir pointing during this maneuver, so target satellites above the Earth Horizon can be tracked.
Delta-H	Target availability depends on the attitude and attitude rates of TRMM.

Table 4.2-8 HGA Usage in Each Control Mode

4.2.4 ACE Operations

The ACE provides an interface for sensor data, processes actuator and EVD commands, and provides an independent SafeHold control mode. ACE software is an interrupt driven system with no operating system. ACE RAM is re programmable on-orbit, however this is not expected to occur since the majority of control functions reside in the ACS processor.

The ACE generates a variety of telemetry in both 2 Hz and 8 Hz modes. Spacecraft event messages are also generated asynchronously as required, and are both downlinked in real-time and passed to the FDS Data Storage for recording in bulk memory. Up to five event messages may be recorded per event message packet, with each message having up to two variable parameters specified. The ACE will only process one ground command per second, and only one command originating from the ACS task per control cycle (two per second). No real-time command activities have been identified which would require a higher command rate to the ACE.

4.2.4.1 Initialization

Two types of ACE initialization are possible, a cold start and a warm restart. ACE cold starts occur when the ACE is powered ON, when the Watchdog timer expires, or upon ground command. An ACE cold start will initialize code and data within the RAM portion of memory in the ACE. Actuator commands are cleared, and the BCRT shared memory is initialized. ACE software is copied from EEPROM to RAM and the ACE will autonomously start execution of the nominal ACE software. An ACE warm start occurs when a cold start is performed, or upon ground command. An ACE warm start re-initializes the ACE software without affecting any code modifications which have been made since the last cold start. An ACE cold start takes approximately 18 seconds to complete, while an ACE warm start takes less than a second and will not interfere with normal operations.

Ground operations must configure the Actuator/Sensor configuration to ACE A (or optionally to ACE B) at initial power ON.

4.2.4.2 ACE Memory Operations

ACE Flight code resides in the 32 KB EEPROM. Since modification of the 32 KB EEPROM is disabled on-orbit all code modifications must be to the 32K low order RAM. Modifying this area presents a problem in that low order RAM is zeroed out during a cold start. This will require the FOT to reload all code modifications to date after a cold start. The following is a generic procedure for loading the ACE.

a. Command ACE B into Boot mode.
b. Load/dump ACE B.
c. Command ACE B to Normal mode.
d. Re configure ACE B as prime.
e. Re configure ACE A as back-up.

f. Load/dump ACE A.
g. Re configure ACE A as prime.
h. Re configure ACE B as back-up.

ACE memory has two regions, bank 0 (RAM) and bank 1 (ROM). Both locations may be dumped, but bank 1 requires that the ACE be commanded to Boot mode before dumping memory. The following is a generic procedure for dumping ACE memory.

a. If bank 1 is requested, switch the ACE to be dumped to Boot mode.
b. Specify the dump start address, bank number, and dump length.
c. Transmit the dump command.

ACE memory load commands follow the one command per second restriction for all ACE commands. The command metering rate through ground stations of 2000 bps will violate this restriction for ACE command loads, given the size of ACE load commands. Loading ACE memory through ground stations involves multiple failures. However, if the need arises, the FOT will ensure that only one ACE load command is stored in each Nascom block, which will reduce the effective uplink rate.

4.2.4.3 ACE/FDS Communication

The FDS bus controller acquires data from and transmits data to the ACEs using a Military Standard 1773 fiber optics interface to each ACE Bus Controller/Remote Terminal (BCRT) shared memory. The ACE shared memory is mapped at ACE software boot time (cold start). The ACS Processor of the FDS operates as the Bus Controller and each ACE operates as a Remote Terminal. The FDS receives data from ACE A first, followed by ACE B. The fiber optics interface is a time multiplexed message (data packet) transfer system. Bus scheduling is established to prevent simultaneous access to the ACE shared memory by the ACE software and the ACS processor. The ACE is locked out of shared memory during BCRT shared memory accesses.

The ACE collects and stores packet data in shared memory without the CCSDS packet headers at a specified transfer address. The FDS software reads the ACE data over the 1773 bus, creates the CCSDS packet headers for all ACE telemetry data packets, and routes the data to a specified destination. The FDS may perform a 1773 bus retry on a read error of packets.

Unpredictable results will occur if both the ACE software and FDS BC access shared memory simultaneously. ACE hardware monitors BCRT and 8085 signals, which are used to access the shared memory, and simultaneously occurring signals are latched as a detected shared memory violation. ACE software polls this latch to detect if shared memory violations have occurred and includes this information in telemetry.

It is possible to experience command problems with the ACE due to SEUs occurring on the fiber optics data bus. A data transmission protocol is employed where a data message is sent and a status message is sent back. If no status message returns, then a retransmission is sent. Only one retransmission is tried. If an SEU occurs on the returned status message it is possible for the command to be sent twice. Other scenarios (TBD) may cause commands never to reach the ACE.

The ACE expects shared memory to be read at specific times based on the ACS timing table. If a message retransmission occurs while the ACE is attempting to access this memory, a severe failure could occur. SEUs are expected at a rate of one every 4.66 days on the fiber optics data bus. The FOT will monitor telemetry to determine if a problem has occurred.

ACE data, and therefore all sensor and actuator data collected by the ACE, are packetized by the FDS task on the ACS processor. If this processor goes down, no ACE data will be recorded or downlinked in real-time telemetry.

4.2.4.4 Timing Signal Processing

The ACE receives a 1 Hz timing signal from either of the two available SDSs. If the timing signal is not received, or 1 Hz synchronization is disabled, the ACE will run on its own internal clock. Eventually, the ACE internal clock and the master clock will drift, which could lead to shared memory conflicts. Access to this area of memory is strictly based on a timing schedule.

4.2.4.5 ACE SafeHold

ACE SafeHold provides a power and thermally safe attitude for an indefinite period. Spacecraft attitude is maintained such that the Sun-line is in the spacecraft's XY-plane and 16.5° from the +X axis in the -Y axis direction. While in SafeHold, the ACE commands the SAs to the indexed position and computes RWA and MTB commands. SafeHold is implemented in the ACE microprocessors independent of the ACS processor and control algorithms. Note that the SafeHold control mode differs substantially from the Low Power mode described in section 4.4.5. Being in the Low Power mode does not necessarily entail being in spacecraft SafeHold.

The ACE will enter SafeHold if the ACS "I'm OK" signal has not been received by the ACE for 7 seconds. This can happen in several ways. The "I'm OK" signal may be suspended by ground command. It also can be suspended by a request from the ACS FDC. The "I'm OK" signal may not reach the ACE or in the event of an ACS/ACE communications failure across the 1773 data bus. Finally, in the event of an ACS processor failure or cold restart, the "I'm OK" will not be sent. SafeHold will not be entered in the event of an ACS processor warm restart, or failure of ACE A and B. In all circumstances, SafeHold may only be exited by ground command.

ACE control is determined differently depending on the state of the spacecraft. Table 4.2-9 summarizes actions taken by the ACE for various modes.

State	Action

Spacecraft has just entered SafeHold.	• ACE waits four minutes for the Solar Arrays to be indexed. • Position errors set to 0. Rate nulling only. • CSS set determined.
Spacecraft is in orbit night.	• Orbit night determined when CSS inputs are less than threshold value for one minute. • Position errors calculated using measured angular rates.
Spacecraft is in orbit day, but the Sun is in the -X axis hemisphere.	• "flip" is required. Spacecraft rotation occurs about the +Y spacecraft body axis. • Position error determined from CSS data.
Spacecraft is in orbit day, and the Sun is in the +X axis hemisphere.	• Position error determined from CSS data.

Table 4.2-9 ACE SafeHold Control Actions

The following actions are taken by the ACE upon entry into SafeHold.

a. The bi-level signal to the GSACE and IPSDU is raised.
b. The IRU is commanded into the high rate mode. The high rate mode ensures system momentum calculations are as accurate as possible, and provides the best information in telemetry.
c. The torque command filter is enabled.
d. Position errors and integrated position errors are zeroed.
e. Default CSS set is chosen by each ACE. The default set to each ACE is selected while the SAs are being indexed. A timer (four minutes) is used to determine that the Solar Arrays should have reached index position. Once the timer has expired, the CSS set is selected according to which SA reached the closest to the index position.
f. The "In Control" ACE assumes control of all RWAs.

Upon entry, the ACE issues the SafeHold bi-level signal to inform the spacecraft that the ACE has assumed control. The GSACE will index the SAs and the IPSDU will notify the instruments. Table 4.2-10 summarizes subsystem configuration in ACE SafeHold.

Component	Configuration
Solar Arrays	Indexed.
HGA	Indexed. Coarse pointing is available if the ACS is in Standby mode.
CERES	Detectors stowed. Instrument powered OFF.
LIS	Instrument powered OFF.
PR	Powers itself OFF prior to the Non-Essential Bus powered OFF.
VIRS	Closes Aperture shutter, closes doors. Instrument powered OFF.
TMI	Instrument powered OFF.

PSDU	Relay to Non-Essential Bus OPEN after 90 seconds.

Table 4.2-10 ACE SafeHold Actions

4.2.4.6 ACE Interrupt Tasks

ACE software is interrupt-driven with three types of interrupts. The BCRT interrupt has the highest interrupt priority, followed by the 8 Hz interrupt and then the 2 Hz interrupt. Each interrupt is associated with a specific set of operations for the ACE software to perform.

The BCRT interrupt is generated whenever the BCRT receives command data over the ACS 1773 data bus. Command data received is copied to RAM to avoid the potential of shared memory conflicts with the ACS processor. As the BCRT has the highest priority, commands are processed immediately. Command types are grouped into Actuator commands, Hardware configuration commands, Software configuration commands, and ground commands.

The 8 Hz interrupt has the next highest priority. Upon receipt, the ACE collects raw sensor data at 8 Hz and stores this data in shared memory for transfer to the ACS processor. An 8 Hz cycle counter is maintained.

The 2 Hz interrupt is the lowest priority interrupt. Upon receipt, the ACE collects ACS housekeeping data and ACE software status telemetry at 2 Hz and stores this data in shared memory for transfer to the ACS processor. Also, if any ACE software status messages have been generated these are stored in shared memory for transfer to the ACS processor and a message counter is incremented for the FDS bus controller.

SafeHold control processing is initiated at the 2 Hz interrupt. SafeHold algorithms are continuously run, however no commands are issued unless the SafeHold mode has been entered.

4.2.4.7 Health and Safety

The ACE maintains a Health and Safety function. The Health and Safety task executes as the lowest priority task and utilizes all remaining CPU idle time. Operations include an ACE Watchdog timer, ACE A to ACE B "I'm OK" checks, and memory checks.

The ACE Watchdog timer reset occurs at the 2 Hz interrupt. Failure to reset the Watchdog timer will result in an ACE cold start. Additionally, every 2 Hz, ACE A sends an "I'm OK" signal to ACE B. In the event that this signal is not received by ACE B in 1.5 seconds, ACE B will configure the ACS actuator and sensor electronics for the B side and ACE B shall assume control. There is no counter operation to fail to ACE A once a transition to ACE B has occurred.

The ACE also computes CRC-based polynomial checksums on sections of low order RAM. Both ACE A and ACE B are checksummed. If the checksum for ACE A does not match the expected value, the "I'm OK" signal is withheld so that ACE B takes over. No control actions are taken if the ACE B checksum does not match the expected value. In both cases event messages are generated. Approximately 2 KB of RAM will be checksummed in 1 second.

4.3 ELECTRICAL SUBSYSTEM

The Electrical subsystem covers the areas of power switching and distribution, command and telemetry routing, discrete telemetry collection, and discrete command distribution. Pyrotechnics, launch vehicle interface, and special test interfaces are also supported by the Electrical subsystem. The components of the Electrical subsystem are the Power Switching and Distribution Units (PSDUs), the electrical harnessing, and optical harnessing. In addition, the Gimbal and Solar Array Electronics (GSACE) contains several PSDU modules for added support as discussed in the following paragraphs.

The TRMM spacecraft contains two separate PSDUs; a Spacecraft PSDU (SPSDU) and an Instrument PSDU (IPSDU). Each PSDU is internally redundant, having a side A and a side B. Despite their names, the IPSDU and SPSDU are not limited to instrument only and spacecraft only interfaces due to the size constraints on the TRMM satellite. Therefore, both PSDUs contain a mixture of functions. The Electrical subsystem provides power to the various subsystems and instruments through the SPSDU and the IPSDU. 28 V is provided to the TRMM spacecraft via redundant power buses. 28 V is the "normal" voltage, however, power supplied to the PSDUs by the Power subsystem can vary from 21 to 35 volts. The Power Bus Interface Unit (PBIU), as part of the Power subsystem, is responsible for distributing the 28 V direct current (DC) from the batteries and SAs to the busses.

The TRMM design contains 3 power busses: Essential, Switched Essential, and Non-Essential. Essential power is used for all spacecraft subsystems that should never be turned OFF, such as the C&DH uplink card, receivers, etc. Switched Essential power is used for subsystems that are normally powered ON, but may be turned OFF in certain situations. For example, half of the Precipitation Radar survival heaters are powered OFF for launch to conserve power. Non-

Essential power is used for all systems that do not fall into the first two categories, and for subsystems that should be turned OFF for Safe Hold and/or Low Power modes. Table 4.3-1 depicts the spacecraft bus configuration in the Electrical subsystem.

ESSENTIAL	SWITCHED ESSENTIAL	NON-ESSENTIAL
SPSDU DCM	SDS S/C processor	CERES
IPSDU DCM	SDS ACS processor	LIS
GSACE DCM	ACE	PR
TAM Electronics	Frequency Standard	TMI
Propellant line Heaters	Clock Card	VIRS
Propellant tank and valve Heater	HGAS main hinge Damper Heater	High Pressure Transducer
FDM, PCM operational Heater	SADDS hinge/damper Heater	Operational Heaters
SDS Uplink Cards	ESA	TMI Survival Heater
Transponder Receiver	IRU	
RWA operational Heater	Power Amplifiers	
DSS electronics	RWA	
Instrument Survival Heaters	Standard Pressure Transducer	
EVD operational Heaters	Precision Pressure Transducer	
Battery operational Heaters	VIRS shutter	
Star Coupler operational Heaters	Catalyst Bed Heaters	
TAM operational Heater	Transponder Transmitter	
ESA operational Heater	Gimbal Module	
PBIU, PSIB Heater	EVD electronics	
PR Survival Heaters	EVD Isolation Valve	
IRU Operational Heater	Thrusters	

Table 4.3-1 Bus Configuration

4.3.1 Power Switching and Distribution Unit

Each PSDU is composed of a combination of module assemblies which are responsible for fusing and distributing power from the Power subsystem, as well as handling any telemetry and command requirements that can not be transmitted over the 1773 fiber optic bus. The modules incorporated into a PSDU (as applicable) are the Divisible (D)-bus Control Module (DCM), Signal Processing Module (SPM), Thermistor Monitoring Module (TMM), Power Distribution Module (PDM), Power Switching Module (PSM), High Power Switching Module (HPSM), Command Signal Module (CSM), and Pyrotechnic Initiation Module (PIM). Each module is fully redundant within itself. The following sections describe the components that make up the SPSDU, IPSDU, and GSACE. Figure 4.3-1 depicts the PSDUs and GSACE power distribution.

The IPSDU contains eight modules and is mainly used for the power and temperature controlling and monitoring of the TRMM instruments. In addition, this unit distributes the SafeHold/Low power signal to the PR, VIRS, and CERES instruments and provides all instrument survival heater power distribution. The SPSDU contains seven modules and provides spacecraft power and pyrotechnic control. It distributes spacecraft survival heater power and provides an on-board sequencer capability. The GSACE contains modules for additional electrical support of all the deployable subsystem's fused and essential power service requirements. In addition, it contains essential power service to the PR survival heaters. Modules assigned to each unit are shown below in Table 4.3-2.

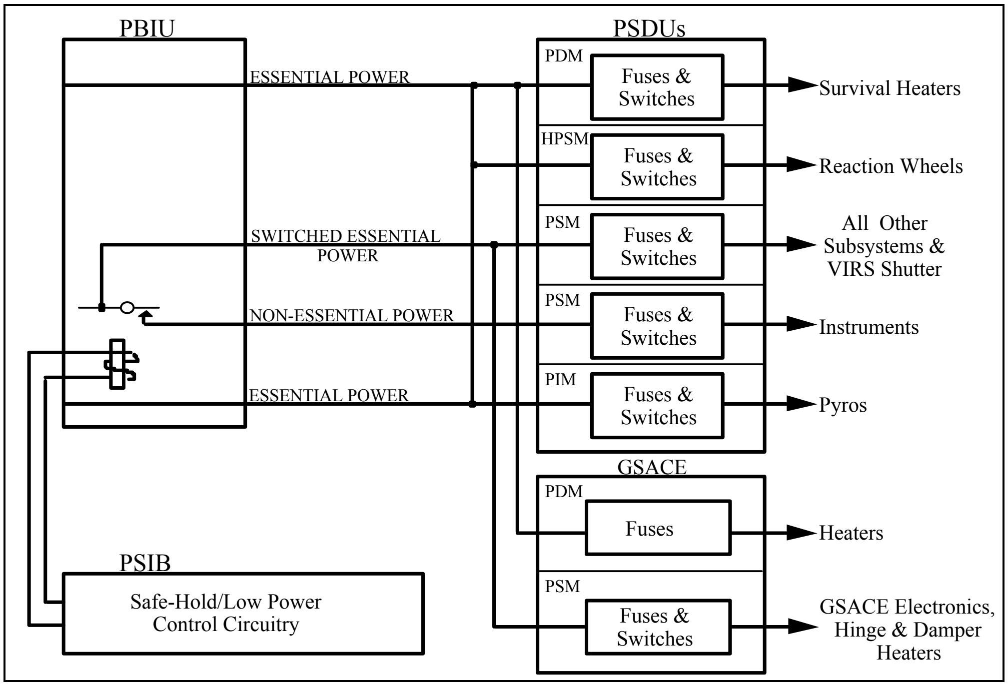


Figure 4.3-1 Power Distribution

Module	Acronym	IPSDU	SPSDU	GSACE
D-bus Control Module	DCM	1	1	1
Signal Processing Module	SPM	2	-	-
Thermistor Monitoring Module	TMM	1	-	-
Power Distribution Module	PDM	1	1	1
Power Switching Module	PSM	2	2	1
High Power Switching Module	HPSM	-	1	-
Command Signal Module	CSM	1	-	-
Pyrotechnic Initiation Module	PIM	-	2	-

Table 4.3-2 Module Distribution

4.3.1.1 D-bus Control Module

The DCM interfaces with the FDS via the 1773 bus and provides control of the PSDU backplane (D-bus). The DCM is the path that all telemetry collected by the PSDU and all commands to the

PSDU passes through. Two modes are supported by the DCM: the Remote Terminal (RT) Command Mode and the Sequencer Mode. The RT Command Mode routes power ON/OFF commands from the FDS over the D-bus. A command from the FDS can be used for powering ON or OFF either the A or B side. The DCM allows a reset signal to be sent to all modules, as well as provides power (+5V, ±15V, +30V) over the D-bus to all other modules. The Sequencer Mode in the SPSDU is responsible for executing launch sequencing commands. The sequencer is initiated automatically by fairing separation and deployment from the launch vehicle and can be terminated by stored commands, 1773 bus commands, or if the sequencer code 'gets lost'. The status of the sequencer will be available over the 1773 bus. The sequencer mode in the GSACE is initiated by receiving the START signal from TMM-A and the ENABLE signal from TMM-B. The signals are sent according to Solar Array hinge potentiometer readings as described in section 4.3.1.6. In addition, the DCM provides output voltage status telemetry of each module. Figure 4.3-2 depicts the telemetry services provided by the DCM and its interface with the SPM and TMM.

4.3.1.2 Power Distribution Module

The PDM provides power distribution for the Essential Bus. This module does not contain any electronics. The PDM receives power input services which are then fused and wired out to up to 18 power services. The capability to switch groups of services and provide termination for the D-bus backplane also exists with this module. The PDM contains relays that will disable some or all of its outputs. However, these relays will only be used in thermal vacuum testing and will remain closed while in flight. It is important to note that if required, these relays can be operated in flight for extremely special circumstances.

4.3.1.3 Power Switching Module

The PSM receives power input services, switches power using 8 relays, and distributes the power to up to 18 power services. Non-essential and switched essential services are distributed by the PSM to its users. Each output is fused via external fuse connectors. The PSM monitors the relay status and current being drawn through each relay and also provides an external reset of all non-essential relays. Each unit in the PSM is cross-strapped to provide redundant services.

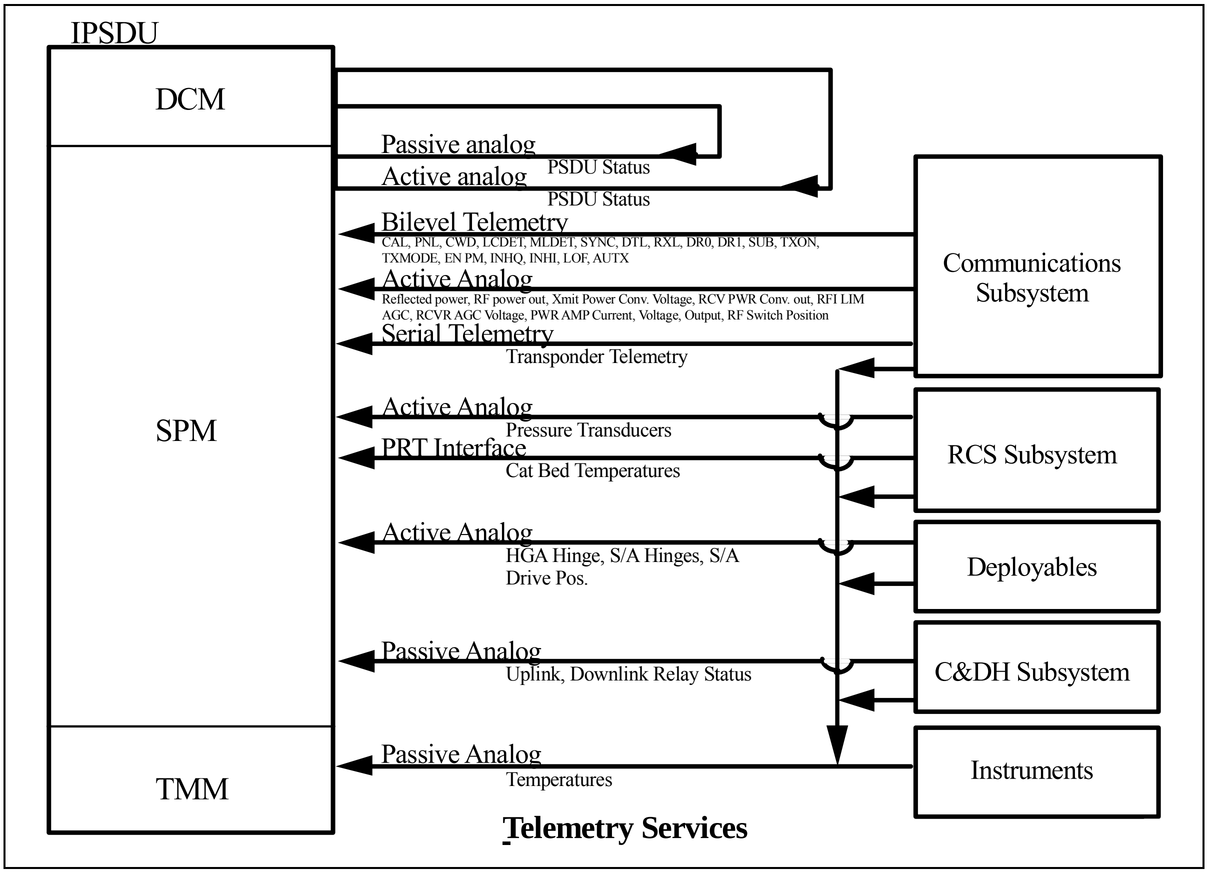


Figure 4.3-2 Telemetry Services

4.3.1.4 High Power Switching Module

The HPSM receives input services from the PBIU and provides power to the four reaction wheels which require switched essential services. The module's circuitry controls the relays and monitors the relay and current status for each service. Each relay can be independently set and reset by the FDS.

4.3.1.5 Signal Processing Module

The SPM is used for monitoring analog or digital signals such as temperature, deployment status, relay status, and operating conditions. This module provides 64 analog input channels. Each channel can monitor voltages between 0 V and 5 V, any resistance, and bilevel signals (0 or 1). In addition, 16 of the channels can receive serial digital telemetry.

4.3.1.6 Thermistor Monitoring Module

The TMM provides the capability to monitor spacecraft and instrument thermistors. It can monitor up to 136 standard thermistors (-25°C to +75°C) and up to 12 Platinum Resistance Thermometers (PRTs) (-25°C to +1700°C). Command signals for the GSACE DCM-A sequencer initiation is additionally provided by the (8 SA pot positions) TMM via 8 SA potentiometer positions. The TMM activates these signals when the +Y Solar Array hinge potentiometers reach their programmed threshold value. The TMM is also responsible for the filtering and distribution of the Safe Hold/Low Power mode signal to up to 12 users.

4.3.1.7 Command Signal Module

The CSM provides command interfaces to the observatory components for functions that cannot be accomplished over the 1773 bus. It provides 62 pulse command lines, each pulse is 28 V and can be programmed with a duration between 1 and 256 ms, which can be used as a logic level, a relay drive interface for ON/OFF control, or other user specific control applications, such as RF switches. Figure 4.3-3 depicts the command services provided by the CSM. In addition, 8 serial digital command lines are provided. This serial interface is used for communicating with the transponders for transponder mode changes, frequency offset, and frequency standard adjustments. Serial commands can only be sent if the CSM in ON. The CSM is also responsible for receiving and distributing the 1 Hz timing signals received from the clock cards. These services are shown in Figure 4.3-4. Figure 4.3-5 depicts the TRMM clock distribution and the cross-strapping of the CSM.

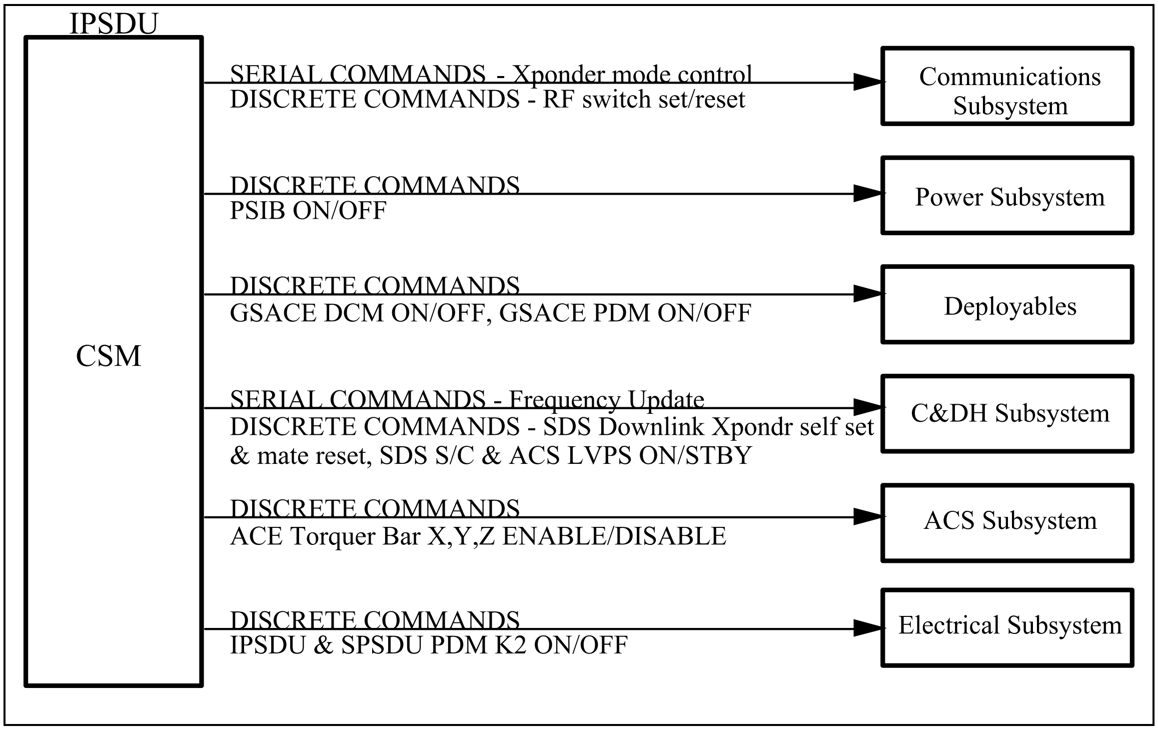


Figure 4.3-3 Command Services

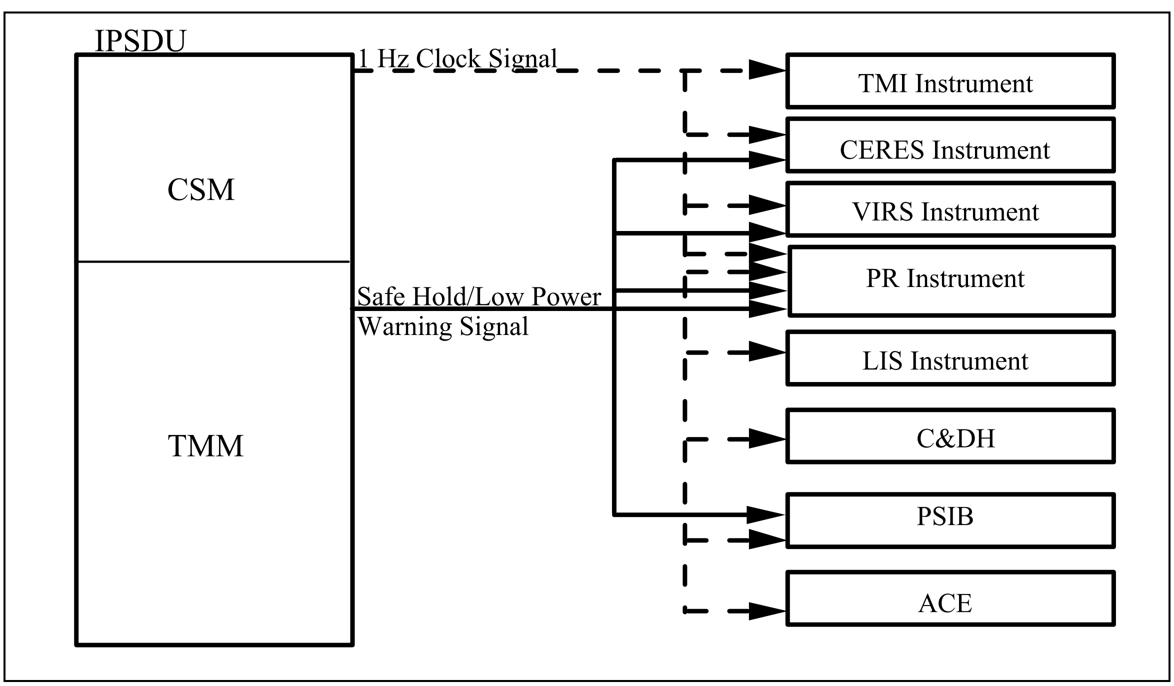


Figure 4.3-4 Special Services

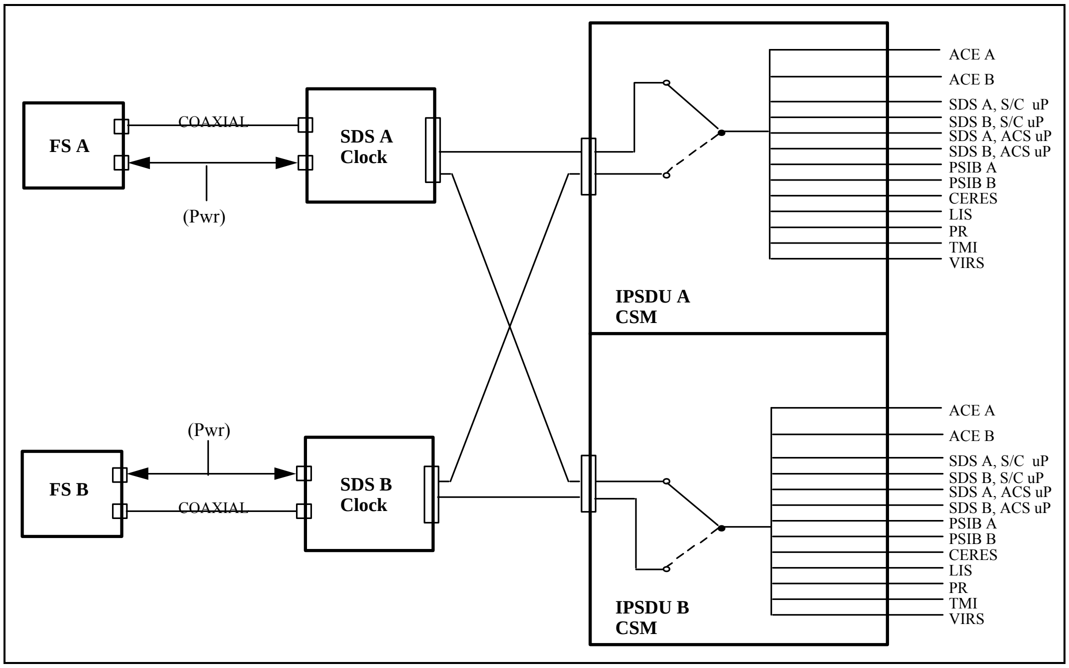


Figure 4.3-5 Clock Distribution and Cross Strapping

4.3.1.8 Pyrotechnic Initiation Module

The PIM provides the capability to safely control the firing of the spacecraft pyrotechnic devices. This module accepts one input service from the Power subsystem PBIU and uses this to power up to 16 pyrotechnic devices. Three commands are required to fire any output service: ARM, ENABLE, and FIRE. The outputs are paired such that they are always fired in groups of two. A single ARM relay is used for all of the services. It is activated by issuing the ARM command and must be set before the FIRE relays are active. The PIM uses a turnaround continuity loop to disable fire capability prior to fairing separation. Figure 4.3-6 shows the sequencing signals and interactions with the PIM.

There is no cross-strapping between the two PIM cards. Full redundancy is achieved by using two Pyros for each pyrotechnic function. Each of the two identical PIM cards will operate simultaneously and power the output services connected to one of the two Pyros. The PIM can be commanded over the D-bus by the Sequencer or by real-time commands. The PIM services are shown in Figure 4.3-7.

Each pyro function, except the RCS pressure valve, is operated by redundant Pyros, with redundant bridge wires. The PIM A side is connected to the primary bridge wires and the B side is connected to the redundant bridge wires. The DCM sequencer will fire the A side of the PIM. In case of sequencer failure, a software sequencer will "shadow" the hardware and fire the B side PIM. PIM commands can also be sent via ground command.

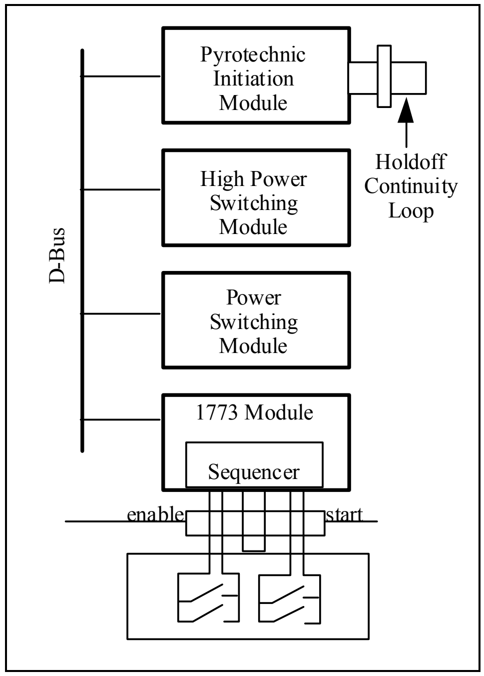


Figure 4.3-6 Sequencing Signals

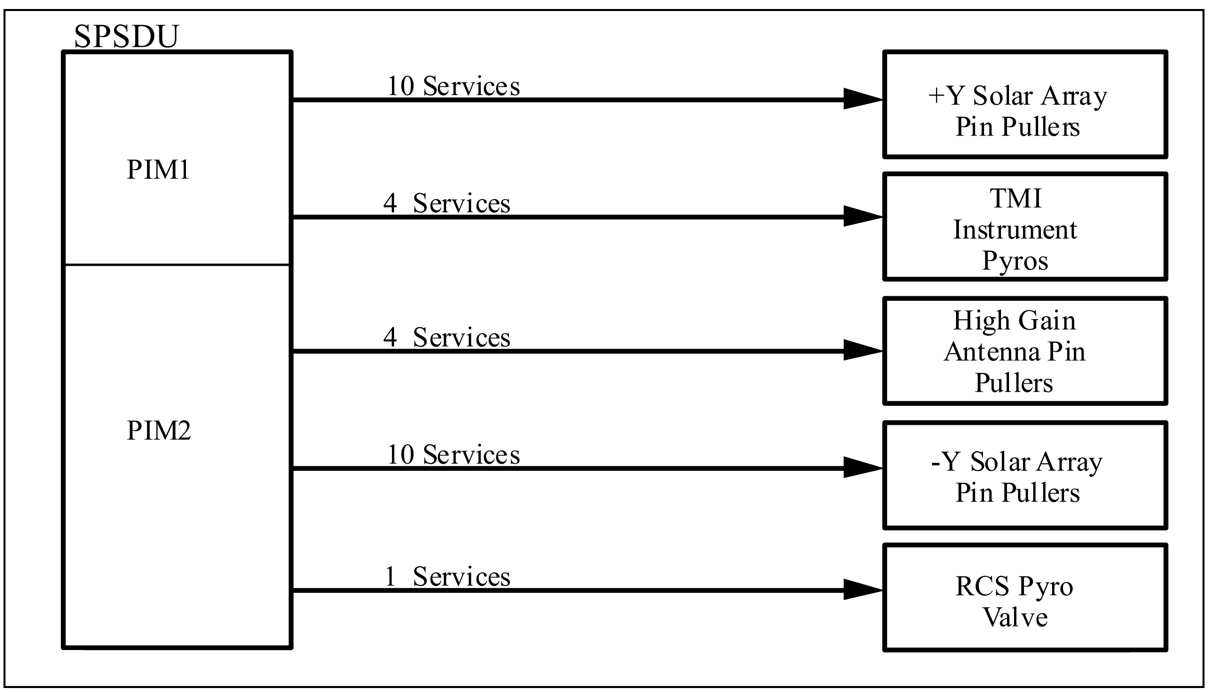


Figure 4.3-7 Pyrotechnic Services

4.3.2 Electrical Subsystem Operations

The Electrical subsystem has no routine ground operations. Normal configuration is PSDU A side being primary. The IPSDU transfers the Safe Hold/Low power signal to the instruments and other users. The SPSDU controls the pyrotechnic operations. In addition, the PSDUs contain the sequencer, which is 'kicked off' after fairing separation.

4.4 POWER SUBSYSTEM

The Power subsystem of the TRMM spacecraft is responsible for providing power for the observatory. An orbital average load of 1100 Watts must be maintained. The Power subsystem must provide support for the mission lifetime. A bus voltage of 28 ± 6 Volts DC, at the PBIU output connector, will be accomplished by the Power subsystem. A bus voltage of 28 ± 7 Volts is expected at the user. The TRMM Power subsystem block diagram is shown in Figure 4.4-1.

The Power subsystem for the TRMM spacecraft consists of the following equipment:

a. Batteries
b. Solar Arrays
c. Power Supply Electronics (PSE)
    1) Power System Interface Box (PSIB)
    2) Standard Power Regulator Unit (SPRU)
    3) Power Bus Interface Unit (PBIU)

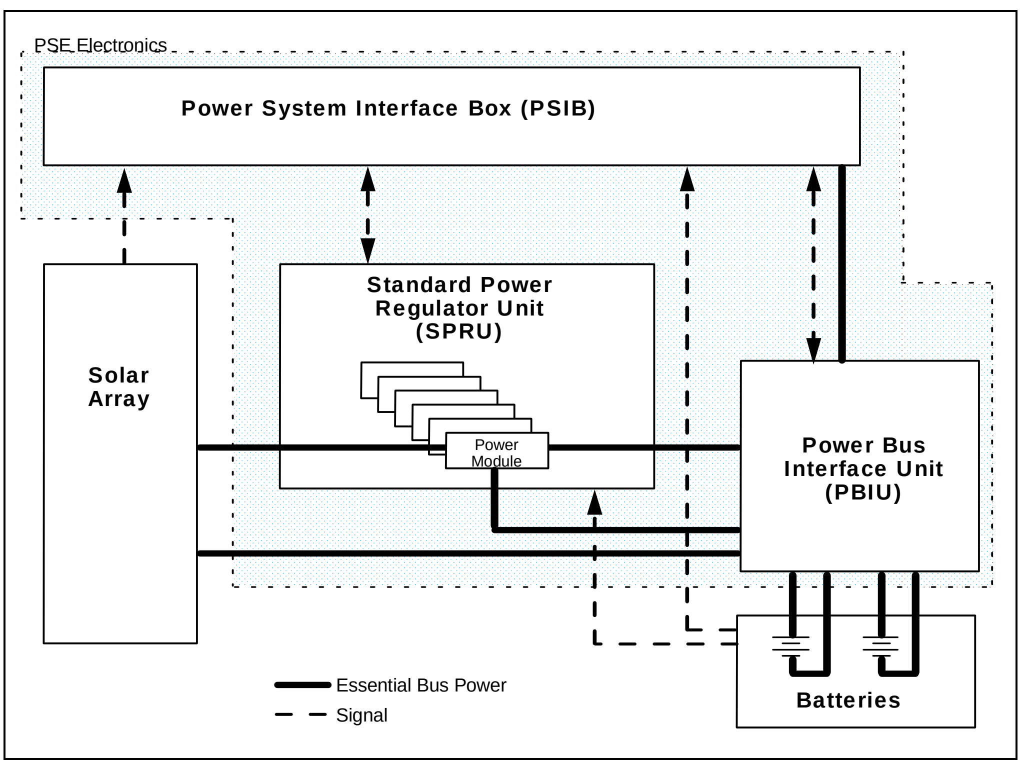


Figure 4.4-1 TRMM Power Subsystem Block Diagram

4.4.1 Batteries

The TRMM observatory is equipped with two 50 amp-hr Super Nickel-Cadmium (NiCd) batteries. The batteries are responsible for providing power to the observatory during periods of shadow. The average eclipse load of 1100 Watts will last for a duration of 26 to 36 minutes with a 25% maximum Depth of Discharge (DOD). The batteries are capable of withstanding a one time 60% DOD. However, TRMM is not expected to see such a discharge, even during launch. The batteries supply power at a nominal voltage of 28 Volts. Each battery has 22 cells connected in series. The thermal operating range is 0°C to 20°C, with a preferred temperature between 5° and 15°. The battery voltage during charge is limited to the maximum voltage for the selected VT level. The maximum and minimum voltages of the battery are dependent on age, temperature, and pressure. Appropriate limits will be established and updated as necessary to take into account battery performance due to these factors.

4.4.2 Solar Arrays

The TRMM observatory is supplied with 2 Gallium Arsenide Solar Array wings, each consisting of two panels. A single panel measures approximately 213 cm X 213 cm. Each panel consists of aluminum alloy face sheets of 0.15 mm thickness. The panels each have 34 strings, containing 68 cells per string. Each cell is approximately 200 µm thick, with multilayers of anti-reflective coating. In addition, cover glass will protect the cells from low energy proton damage while covering the entire active cell area. Each wing is canted 26.5° from the boom as shown in Figure 4.4-2.

Figure 4.4-2 Solar Array Orientation

Each Solar Array is sized to provide an average load of 1100 W between Beta 0° and ± 58.5°. The solar arrays are responsible for providing the observatory with power and for recharging the batteries during the sunlit portions of each orbit. During sunlit portions of the orbit, the solar arrays will continuously track the Sun, in one axis, and will be feathered during periods of shadow. The SAs are driven from ACS algorithms as described in section 4.2. The spacecraft will perform a yaw turn each time the Beta angle crosses 0°. This effectively reduces the range of Beta Angles by half to between 0° and +58.5°, and keeps the shadowing of the solar array by the spacecraft on the +Y side.

4.4.3 Power Supply Electronics

The PSE consists of three boxes: the PSIB, SPRU, and PBIU. The PSE is single fault tolerant with built in redundancies which is required to provide 1100 W of power to the observatory. The PSIB provides the interface between the FDS and the Power subsystem. The SPRU provides power regulation for the Power subsystem. The PBIU contains components for providing power to the various busses.

4.4.3.1 Power System Interface Box

The PSIB is a fully redundant system which is responsible for providing the interface between the spacecraft FDS and the Power subsystem. The PSIB provides a digital command and monitoring unit designed to support the Power subsystem requirements for TRMM. Figure 4.4-3 depicts the PSIB interfaces.

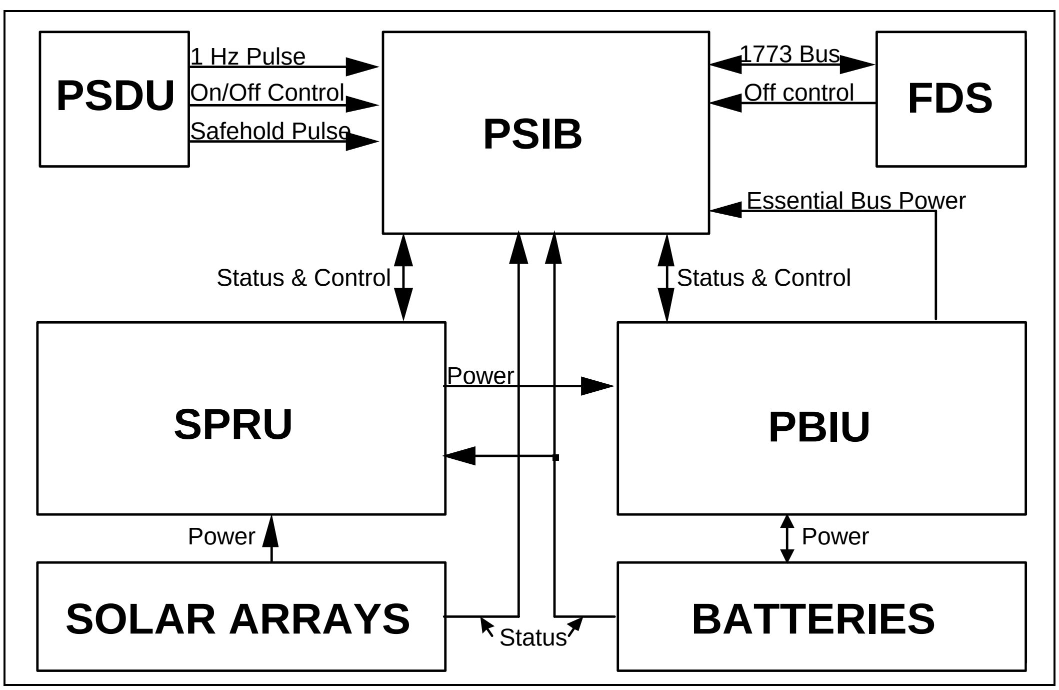


Figure 4.4-3 PSIB Interfaces

The PSIB is a microprocessor-based unit consisting of a processor card and 5 input/output cards connected to the PSIB backplane, as shown in Figure 4.4-4. The processor card supports spacecraft interfaces consisting of the 1773 data bus, 1 Hz pulse, and SafeHold pulse. The input/output cards support the Power subsystem interfaces consisting of the power supply, relay driver, digital I/O, thermistor/PRT, and Analog voltage monitor. The following paragraphs detail the operations of the PSIB and the PSIB flight software.

4.4.3.1.1 Battery State of Charge

The PSIB receives status information from the batteries. From this data, the PSIB calculates the battery state of charge (SOC) and controls the SPRU mode of operation accordingly. If the battery is discharging, the SOC is calculated by equation a. If the battery is charging, the SOC is calculated by equation b.

a. SOC = SOC (last) - Discharge I
b. SOC = (SOC (last)) + (Charge I * (Charge/Discharge Ratio))

The charge/discharge ratio, with a default value of 1.05, can be changed via a 1773 command. It will be displayed in the MOC.

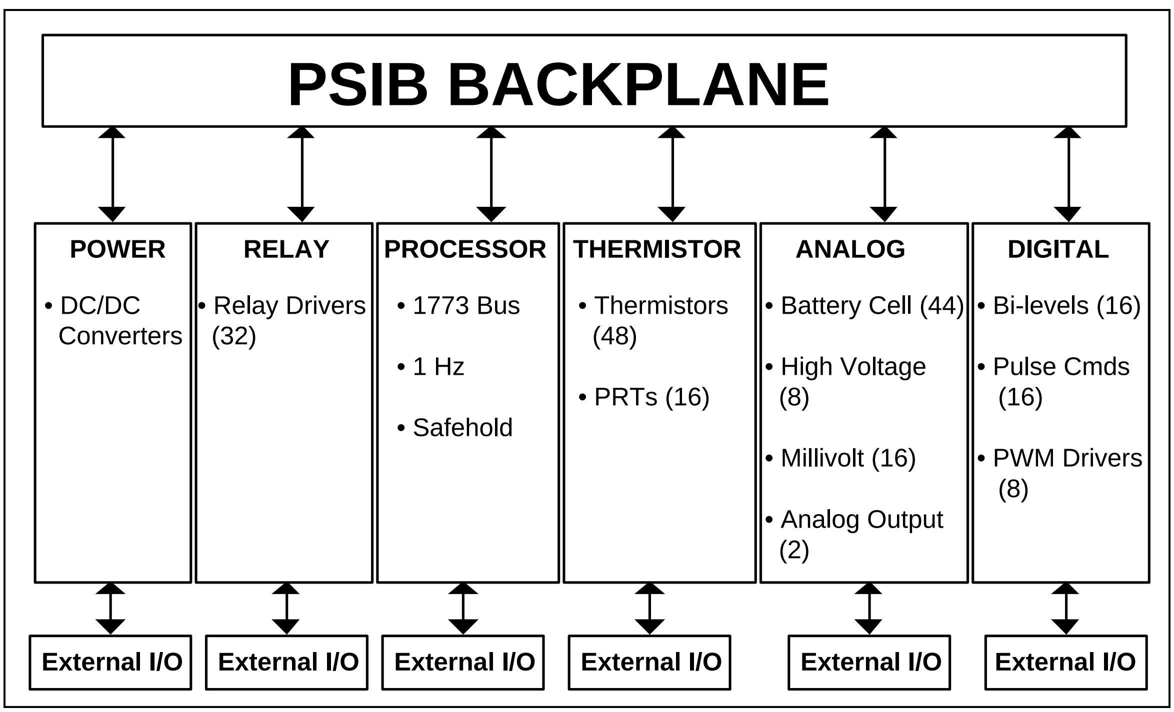


Figure 4.4-4 PSIB Configuration

4.4.3.1.2 Bus Shutdown Conditions

The PSIB is ultimately responsible for shutting down the Non-Essential Bus during low power conditions. The PSIB checks for low battery cell voltage (< 1 V nominal on any cell) and Essential Bus voltage (< 22 V nominal). Each of these limit values are command selectable and must be set higher than the nominal values to allow for the 90 second delay before shutdown. The PSIB will shutdown the Non-Essential Bus 90 seconds after receiving the SafeHold pulse, and after Low Power detection if the SafeHold pulse is not received within 95 seconds. It is important to note, however, that the PSIB must know which side (Side A or B) of the TMM it is expecting the SafeHold/Low Power pulse. If TMM B is operating, rather than the nominal TMM A, then the PSIB must be commanded to look for the signal coming from TMM Side B. In addition, the PSIB's capability to open the Non-Essential bus relay can be disabled. For normal operations, if a safehold/low power pulse is sent out, a TSM will execute an RTS. This RTS will wait 90 seconds and then send commands to open each of the relays that provide power to the instruments. This RTS will also disable the capability for the PSIB to remove power from the Non-Essential bus.

4.4.3.1.3 PSIB Flight Software

The PSIB flight software is a single executable with one interrupt handling routine written in the C programming language. The software is capable of receiving loads to RAM and performing dumps from RAM or EEPROM. Every 5 minutes, the PSIB flight software performs memory checksums and will reboot the PSIB if an error is detected. The PSIB flight software can process up to 20 commands per second from the FDS and will generate packets at a rate of at least one packet per second. Telemetry packets will be time tagged, and will include 4 bits of subseconds. The PSIB flight software will generate five kinds of telemetry packets. It will generate normal and housekeeping packets using stored PSIB channel data. It will also generate memory dump, fast, and diagnostic packets upon request. The purpose and frequency of each packet type is detailed in Table 4.4-1.

Packet Type	Frequency	Use/Data
Normal	1 packet every second	Data for Essential power values (voltages and currents of Buses, SAs, and Batteries)
Housekeeping	1 packet every 30 seconds	Data for thermistors, PRTs, Cell voltages, full and half voltages, thresholds, command counter
Fast	1 packet every 30 seconds (Upon request)	Data for 2 selectable PSIB parameters are sampled 10, 100, or 1000 times per second for debug purposes
Memory Dump	1 packet every 30 seconds (Upon request)	PSIB Memory Dump Data
Diagnostic	1 packet every 30 seconds (Upon request.)	PSIB diagnostic data which contains divider, mask, and status of cards

Table 4.4-1 PSIB Telemetry Packet Types

A memory load/dump capability is provided, however, changes to the PSIB flight software are not expected to be made once on orbit. For the control of load uplinks, a capability will be provided in the MOC to compare the contents of each memory address or table location in the dump image to the corresponding load image. For PSIB memory dumps, the dump data will be identified by APID. Once a load is verified and all quality and error checking requirements have been completed, the dump file will be made available for electronic transfer to the On-board Computer Software Tools (OST) workstation.

4.4.3.2 Standard Power Regulator Unit

The SPRU provides Solar Array power regulation and battery charge control for the Power subsystem. The SPRU supplies the Power subsystem with the following four autonomous modes of operation: Peak Power Tracking Mode, Voltage Limit Mode, Constant Current Mode, and Standby Mode. Mode changes to the Constant Current Mode are controlled by the PSIB and are dependent upon the calculated battery state of charge. The SPRU will autonomously transition into the Peak Power Tracking Mode whenever the bus load demands and battery charging requirements exceed the available power from the Solar Arrays. In this mode, the six SPRU power modules will draw maximum power from the Solar Arrays and will distribute power between the load and batteries. Any remaining power will be provided to the batteries. The SPRU will autonomously transition to the Voltage Limit Mode any time the voltage or temperature limit established by the selected Voltage/Temperature (V/T) curve (see figure 4.4-5) is reached on either battery. The V/T limits are changed via ground command sent to the PSIB. The PSIB then sends a pulse command to the SPRU. During normal operations, the setting of the V/T level should remain the same throughout the mission. V/T levels will be set to maintain battery charge to sustain normal spacecraft loads. In this mode, the V/T control circuitry becomes effective and current being drawn from the SAs is reduced. This results in decreased SPRU input power and output current. The V/T controllers for battery 1 and battery 2 are OR'd together resulting in parallel battery charge controlled by whichever battery hits the V/T limit first.

The PSIB commands the SPRU to Constant Current Mode in order to control total battery charge current. One of seven command selectable constant charge levels (0.75, 1.5, 3.0, 6.0, 12.0, 24.0, and 48.0 Amperes) will be selected. This mode is responsible for restricting overcharge of the batteries. If the batteries are determined to be fully charged, the SPRU will be commanded to operate in constant current mode at either 0.75, 1.5, or 3.0 Amperes. Constant Current Mode can also be used when first entering sunlight in order to limit high battery charge currents. In this method, one of the higher constant charge levels will be used (either 6.0, 12.0, 24.0, or 48.0 amperes.) The SPRU autonomously transitions into Standby Mode when the Solar Array voltage falls below 44 V. In this configuration, command logic and associated power supplies in the SPRU are supplied by battery power. The SPRU remains in Standby mode until the Solar Array voltage exceeds 53.5 V.

Figure 4.4-5 V/T Control Curves

4.4.3.3 Power Bus Interface Unit

The PBIU directs power to the Essential and Non-Essential power Busses and disconnects the batteries from the power Busses when required. The Non-Essential Bus will be disconnected from the batteries when an under-voltage or over-current situation occurs. If any battery cell voltage drops below 1 V or if the Essential Bus voltage drops below 24 V, an under-voltage condition exists. These conditions will be sensed by the PSIB. The PBIU will contain the spacecraft power ground to which all primary power will be returned. The PBIU will be responsible for measuring ground currents through the use of a low resistance shunt between the primary power return and the spacecraft structure. The voltage measured will be passed to the PSIB. The PBIU also contains sensors which measure bus loads and battery performance, although these sensors are monitored by the PSIB. The PBIU is also responsible for supporting on-orbit reconditioning of the battery via ground command.

4.4.4 Power Subsystem Operations

There are four types of Power subsystem operations: Peak Power Tracking of Solar Array output, Battery Charge Control, Battery Reconditioning, and PSIB Memory Load/Dump operations. These operations are detailed in the following paragraphs. PSIB Memory Load/Dump operations were discussed in section 4.4.3.1.3.

Peak Power tracking of Solar Array output occurs continuously during operations, as does Battery Charge Control. Constant Current Charge control is initiated by the PSIB. The PSIB monitors battery current status information. From this data, the PSIB calculates the battery state of charge and controls the SPRU mode of operation as detailed in section 4.4.3.1.1.

The nominal Power subsystem configuration consists of having both batteries connected to the main bus. The two batteries will be charged and discharged in parallel. Normal operations have two primary methods of battery charging. The first method is Voltage/Temperature (V/T) control and the second is Ampere Hour Integrator (AHI) and current control.

In the V/T Control method, the battery with the highest temperature controls the charging of both batteries. The V/T controllers for each battery are OR'd together. The solar arrays charge the batteries with all power not used by the spacecraft load. When any battery reaches the selected temperature compensated voltage limit, the V/T controller reduces the power being drawn from the solar arrays.

The AHI method operates while in the SPRU Constant Current Mode. The AHI counts the total battery charge and discharge. When the AHI recharge count reaches one of three ground selected constant charge levels, the battery charge is switched to a low rate trickle charge mode. The Constant Current Mode is responsible for restricting overcharge of the batteries.

The Power subsystem supports on-orbit battery reconditioning. Battery reconditioning is permitted by removing one of the batteries from the bus, completely discharging it, and returning it to the bus for recharging by placing the battery in Normal Mode. Reconditioning is done via ground commands.

4.4.5 Failure Detection Functions

The PSE failure detection logic is responsible for monitoring the voltages and currents for out of limits conditions. Undervoltage conditions, as mentioned in section 4.4.3.1.2, will trigger a low power configuration. The PSDU will be responsible for sending out the low power pulse. The pulse is sent to all systems relying on the Non-Essential Bus. The PSIB waits 90 seconds after receiving the pulse, then sends a command to the PBIU to open the power disconnect relay for the Non-Essential Bus. As mentioned in 4.4.3.1.2, this will only occur if the RTS to open the relays to the instruments do not execute properly.

4.5 RADIO FREQUENCY COMMUNICATIONS SUBSYSTEM

The Radio Frequency (RF) Communications subsystem provides the interface between the TRMM spacecraft and the Ground Data System (GDS). The primary ground system link is through the Space Network (SN) Tracking and Data Relay Satellite System (TDRSS). The RF subsystem will provide forward and return link communications services for telemetry, command, and tracking operations. For contingency operations, transponder telecommunications can be achieved with the Ground Network (GN) and the Deep Space Network (DSN).

The RF Communications subsystem for the TRMM spacecraft consists of redundant equipment strings (except for the High Gain Antenna and power amplifier) and RF Combiner which include the following components:

a. High Gain Antenna (HGA) with dedicated Antenna Pointing System (APS)
b. Omni-Directional Low Gain Antennas
c. NASA Standard Second-Generation TDRS User Transponders
d. RF Switches
e. Solid State Power Amplifier
f. Interconnecting RF cables
g. Diplexers
h. RF Combiner (Includes directional and hybrid couplers)
i. Band Reject Filters (BRF)

A block diagram of the TRMM RF Communications subsystem is shown in Figure 4.5-1.

4.5.1 TRMM RF Antennas

The TRMM spacecraft is equipped with a HGA and gimbal assembly, and two Omni antennas. The HGA will provide the primary means of communications for science data downlink, health and safety verification, and command uplink. The Omni antennas will provide communications during the initial spacecraft acquisition and early orbit operations checkout phase, and backup communications during Emergency/Contingency operations.

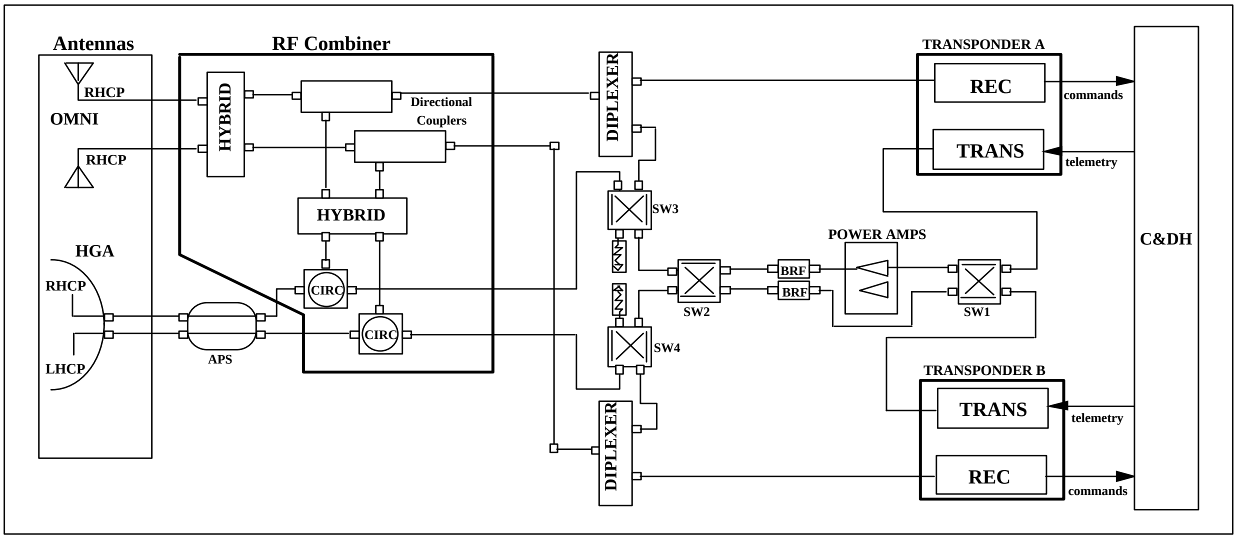


Figure 4.5-1 RF Communications Subsystem Block Diagram

4.5.1.1 Omni Antenna

The TRMM Omni antenna consists of two dipole antennas coupled through a 3 dB 180 degree hybrid which produces a sum and difference signal. Each Omni antenna element is diametrically mounted on opposite sides of the spacecraft. Both antennas are Right Hand Circularly Polarized (RHCP) and have near-hemispherical coverage. The Omni antennas will be utilized to downlink housekeeping telemetry at a rate of 1 or 1.5 Kbps on the Q-channel and fill data at a rate of 1 Kbps on the I-Channel via TDRSS. The TDRSS command uplink rate will be 500 bps or 1000 bps. Once the HGA is deployed and operational, the Omni antennas will only be used for emergency/contingency operations with either the SN, GN, AGO, WFF, or DSN, and to obtain TCXO Center Frequency measurements for Transponder-B. Characteristics of the Omni antenna are summarized in Table 4.5-1.

Characteristic	Omni	Remarks
XMT Frequency (MHz)	2255.502120 MHz	
RCV Frequency (MHz)	2076.941530 MHz	
Polarization	RHCP	
VSWR	< 1.5:1	RCV/XMT
Coverage	nearly spherical for the Omni	
Gain:	0.0 dB	XMT
	1.0 dB	RCV
Location	+X (UISP) -X (lower propellant tank)	

Table 4.5-1 Omni Antenna Characteristics

4.5.1.2 High Gain Antenna System

The High Gain Antenna System consists of a parabolic dish, and a gimbal drive electronic control assembly, with a rotation of 90° in each of the two axes that enables the antenna to point over a field of view. The polarization of the HGA is command selectable between Left-Hand Circular Polarized (LHCP) and Right-Hand Circular Polarized (RHCP). Operationally, the HGA will be LHCP. The Attitude Control System (ACS) microprocessor outputs pointing vectors to command the Antenna Pointing System (APS) to point the HGA to whichever TDRS is selected. Table 4.5-2 provides the characteristics of the HGA.

Stowed for launch, the HGA is mounted on a deployable boom which will be activated through the SPSDU sequencer after spacecraft separation. Once locked into position, antenna boom operations are complete. The HGA will provide almost complete hemispherical coverage (180°), except for keyholes along the Y-axis. The location of the keyholes are such that TDRS coverage is never available at those angles. Figure 4.5-2 depicts a simplified drawing of the HGA and Omni antennas , with respect to the spacecraft.

During normal mission operations, the HGA will be used for downlinking of telemetry to the ground. The FOT will routinely schedule 18 to 19 20-minute real-time supports per day (one every orbit). Command loads will be uplinked to the S/C Stored Command Processor (SCP) to properly configure the RF subsystem for each real-time support. During these supports, real-time housekeeping data will be downlinked via the TDRS I-Channel at a rate of 32 Kbps, and stored commands will be issued to initiate solid-state recorder playbacks at a rate of 2.048 Mbps, via the TDRS Q-Channel.

Characteristic	HGA	Remarks
XMT Frequency (MHz)	2255.502110 MHz	
RCV Frequency (MHz)	2076.941520 MHz	
Polarization	LHCP	Command Selectable to RHCP
VSWR	< 1.5:1	RCV and XMT
Coverage	Hemispherical	Gimbal drive electronic control assembly
Gain:	27.5 dB	XMT
	22.8 dB	RCV
Location	End of Boom -Z axis; located on top of S/C.	
Physical Dimensions	Diameter - 1.323 meters Height - 0.587 meters	

Table 4.5-2 HGA Characteristics

After each real-time support, the RF subsystem will terminate data transmission, and the HGA will be commanded to the feathered position (pointed in the +Z direction), via stored commands from the spacecraft SCP, to minimize the effects of atmospheric drag. The HGA will always be commanded to the feathered position after each real-time support (via stored commands).

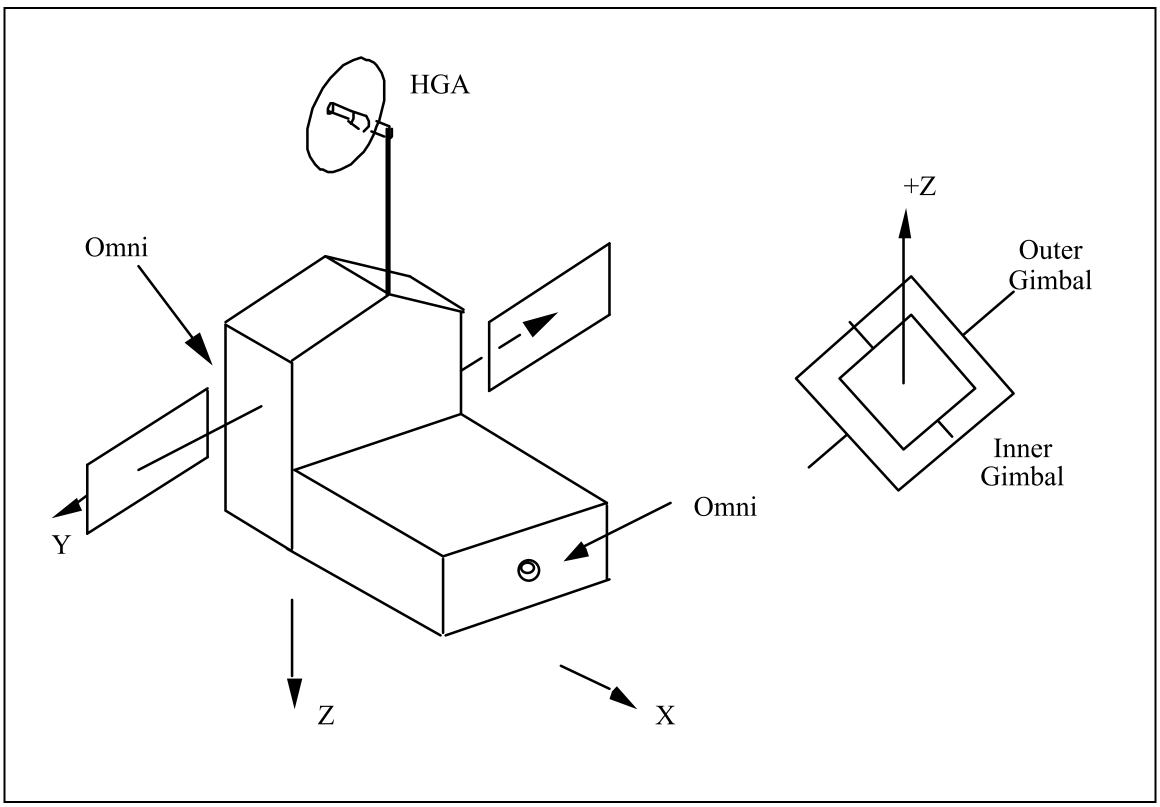


Figure 4.5-2 TRMM Antenna Locations (Simplified)

4.5.2 Gimbal Solar Array Control Electronics

The GSACE (although not a component of the RF subsystem), provides a two-axis gimbal antenna control system, that allows the HGA to track a TDRS. The FOT will uplink the TRMM Extended Precision Vectors (EPVs) daily, and the 4 TDRS EPVs every week to the ACS processor. The ACS processor will propagate the EPVs in order to maintain knowledge of TDRS locations. Prior to each real-time support, stored commands will be issued to initiate tracking of the scheduled TDRS. The GSACE will determine where the commanded TDRS is located, and slew the antenna to point to that location. The FOT will always schedule this slew sequence to occur 4-minutes prior to the scheduled real-time event. This four-minutes is the worst case time required to slew the HGA from the feathered position down to either horizon. Once the HGA reaches the commanded location, the antenna will track the scheduled TDRS until commanded to stop tracking.

4.5.3 Transponder Operations

The TRMM design includes two redundant NASA Standard (Second-Generation) TDRSS User Transponders to provide data transmission, command reception, and ranging signals. The transponders have the capability to receive and transmit S-Band signals compatible with both TDRSS and GN/DSN networks. The transponders also provide a ranging capability through the SN for both one-way and two-way tracking services. Ranging will not be provided via GN/DSN. Each transponder has a RF output of 5.0 Watts and TDRSS Q/I channel power ratio of 4:1. Finally, both transponders will be cross-strapped for redundancy to allow communications via either the Omni or HGA. Note: both receivers are always ON and receive signals from either the HGA or Omni antenna.

Operationally, the transmitter of each unit will be assigned to an antenna and will not be interchanged for normal operations. Transponder A will be configured to the HGA and will be designated as the HGA Transponder. Likewise, Transponder B will be configured to the Omni and will be designated as the Omni Transponder.

4.5.3.1 Receiver Operations

The receiver portion of the transponder accepts an uplink signal from either TDRS or GN/DSN, demodulates the commands, and then forwards them to the Uplink Card for routing to the final destination in the spacecraft. Figure 4.5-3 shows the interface between the RF Communications subsystem and the C&DH. The nominal receive paths for the Communications subsystem are illustrated in Figures 4.5-4 and 4.5-5.

Both receivers are always powered from the Essential Bus and will accept an uplink signal from either a GN/DSN or TDRS forward link source. In the TDRS mode, a long and short Pseudorandom Noise (PN) codes will be received. Once the receiver is locked onto a signal the other signal will be locked out.

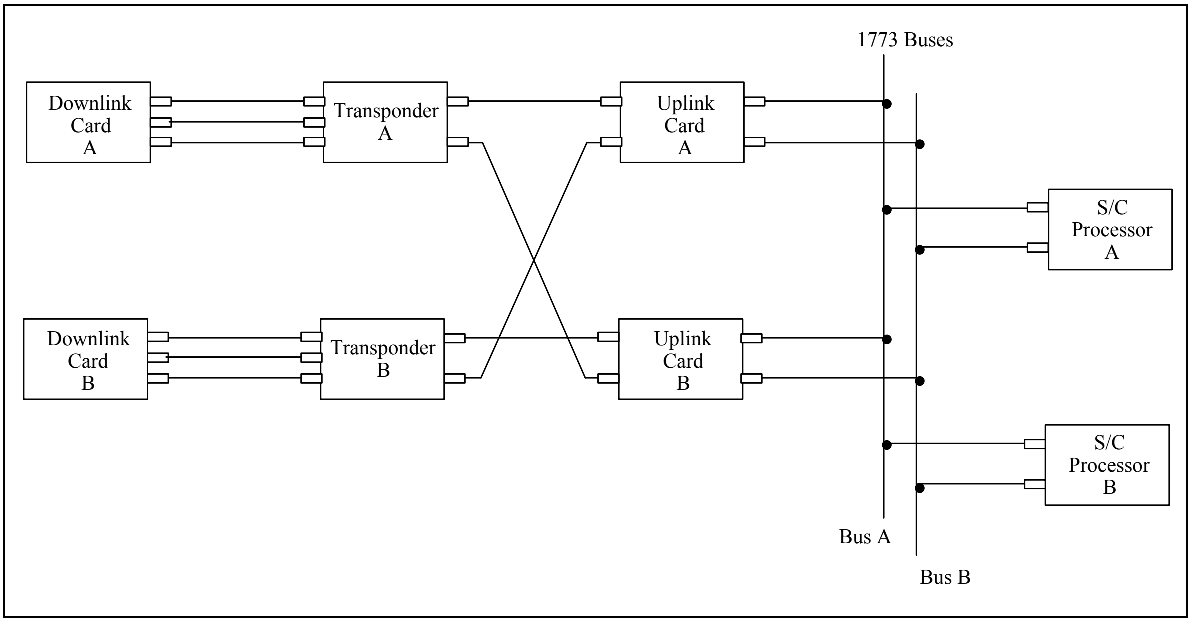


Figure 4.5-3 RF Subsystem to C&DH Interface

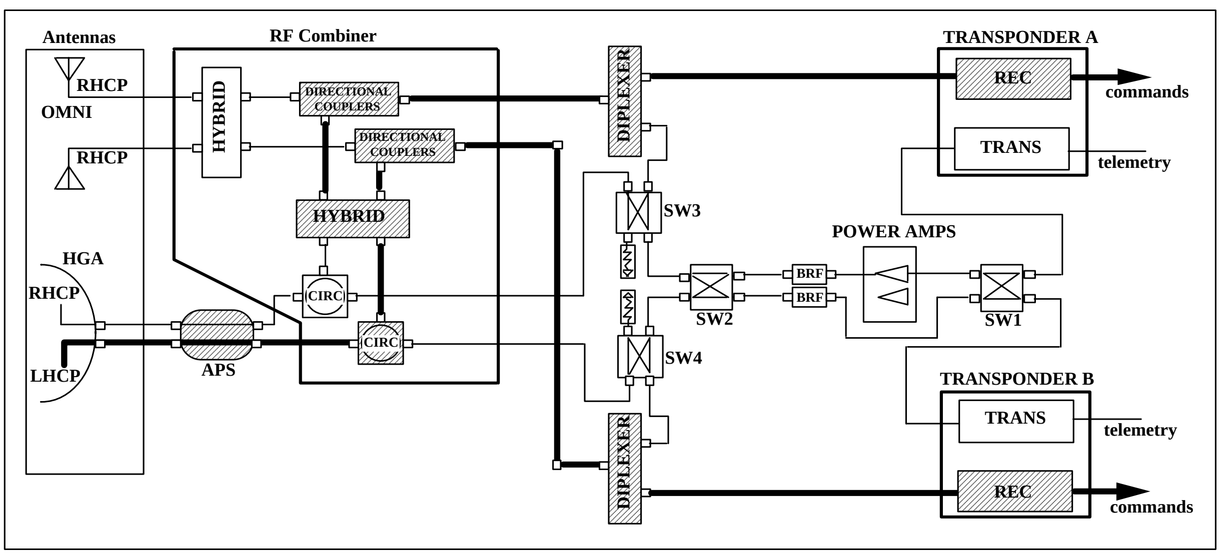


Figure 4.5-4 HGA Receive Path

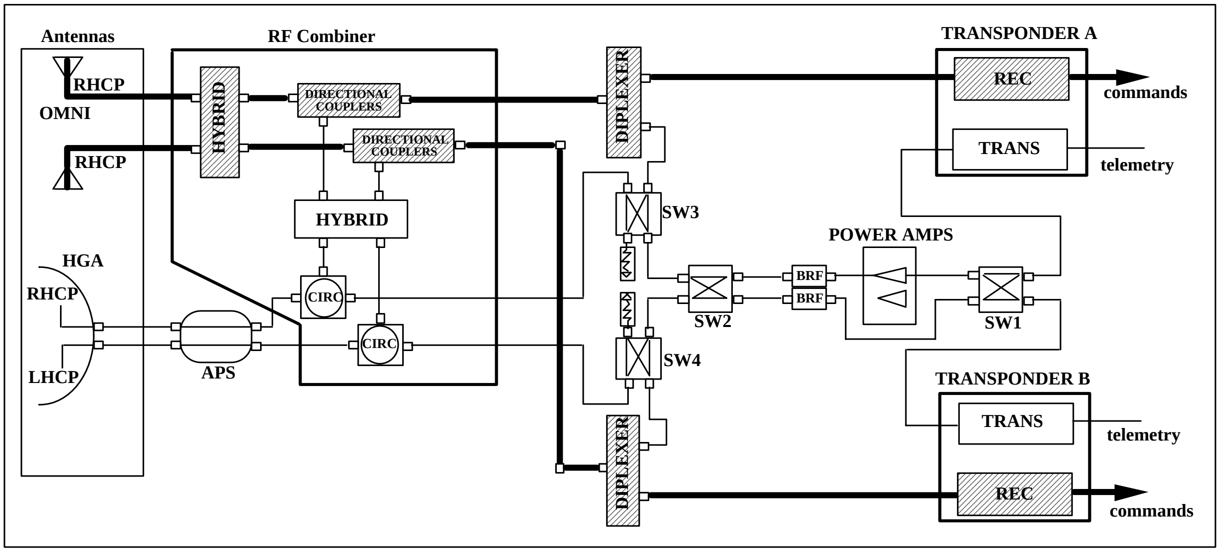


Figure 4.5-5 OMNI Receive Path

4.5.3.1.1 SN Forward Link Operations

TDRS will provide for the transmission of digital command data to the TRMM using the SSA forward link service. This link provides command data to the TRMM at the rate of 500 bps or 1000 bps (nominal rate). Ranging PN codes are transmitted simultaneously with commands to the TRMM spacecraft.

Command data in NRZ-L format is modulo-2 added asynchronously to the command channel Pseudorandom noise (PN) code. The command channel PN code and the range channel PN code are then used to Quadrature Phase Shift Key (QPSK) modulate the forward link carrier. The command channel to range channel power ratio after carrier modulation is 10 dB.

The Second TDRS Ground Terminal (STGT) uplinks the commands at Ku-band to TDRS. In the process, the carrier frequency is Doppler compensated for TDRS-to-TRMM Doppler shift. The capability exists to inhibit Doppler compensation of the forward link carrier during coherent supports (two-way Doppler measurement).

TDRS will then coherently down-convert the signal to 2076.94 MHz and then forward to TRMM. On-board TRMM, the transponder PN receiver acquires, tracks, and despreads the received signal. The short command channel PN code is acquired first by a PN code despreader. The RF carrier is then recovered. The long range channel PN is acquired next. Prior to declaring acquisition on the range code, a test for false lock on a Multipath signal is performed. After signal acquisition, the range channel PN code epoch is routed to the transponder transmit portion to provide synchronization for the return link PN code for the coherent mode of operations.

The recovered carrier is routed to the transmit portion of the transponder to provide turnaround carrier required for the coherent return link operation. The baseband command data is provided to the bit synchronizer, which recovers the data clock and restores the data to a digital NRZ-L signal. The NRZ-L signal is then forwarded to the FDS.

4.5.3.1.2 GN/DSN Uplink Operations

The GN/DSN will provide for the transmission of digital command data to the TRMM using a S-Band uplink service. This link provides command data to the TRMM at the rate of 2000 bps. The commands will be encoded in the NRZ-L format. The formatted command will be used to PSK modulate the 16 KHz subcarrier.

The uplinked signal will be received by the transponder where the Phase Modulated signal will be demodulated to provide the baseband signal. The 16 KHz baseband subcarrier will then be demodulated and passed to the onboard command detector from which command data in the NRZ-L format will be forward to the FDS.

4.5.3.2 Transmitter Operations

The transmitter portion of the transponder transmits a downlink signal to either TDRS or GN/DSN. The nominal transmit paths for the Communications subsystem are illustrated in Figures 4.5-6 and 4.5-7.

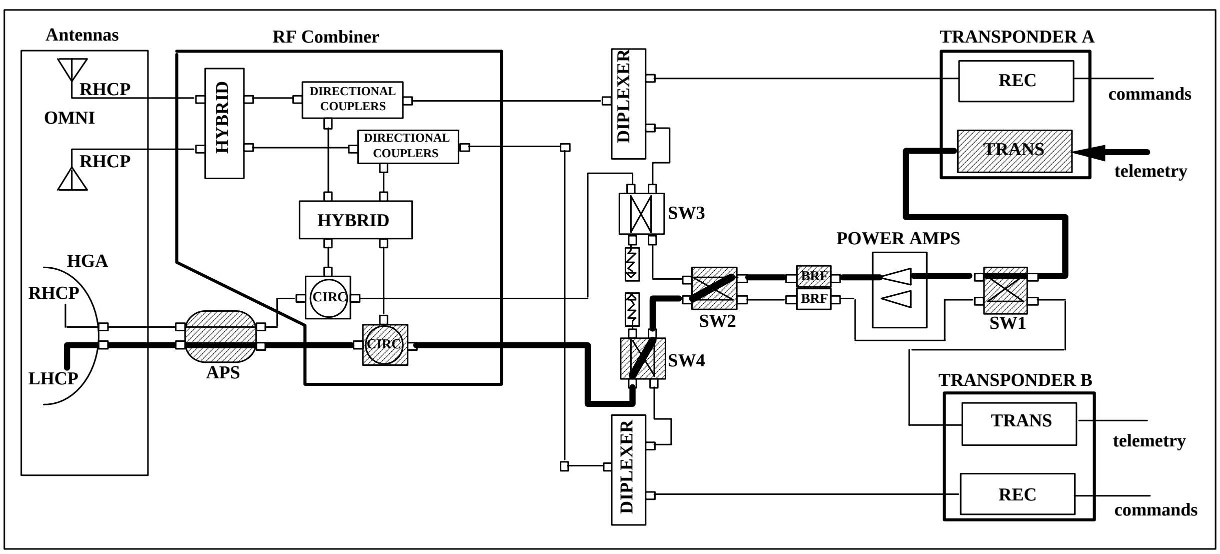


Figure 4.5-6 HGA Transmit Path

4.5.3.2.1 SN Return Link Operations

The S-Band Return link is used to send real-time telemetry and Solid State Recorder playback data from TRMM to the TDRSS. This link is also used for determining range and range rate of the spacecraft. The link is operated in one of two modes: the coherent mode (Data Group 1, Mode 1) and (Data Group 1, Mode 3), or the non-coherent mode (Data Group 1, Mode 2).

The TRMM C&DH design includes several command selectable encoding schemes. Nominally, the real-time and playback data will be Reed-Solomon encoded and rate 1/2 convolutionally encoded. The I- and Q-channel PN codes modulate (PSK) the Quadrature phase components of the carrier. The Q-channel to I-channel power ratio after modulation is 3 to 1. Figure 4.5-8 provides a functional diagram of the on-board encoding schemes implemented (within the FDS Downlink Card) to support the mission.

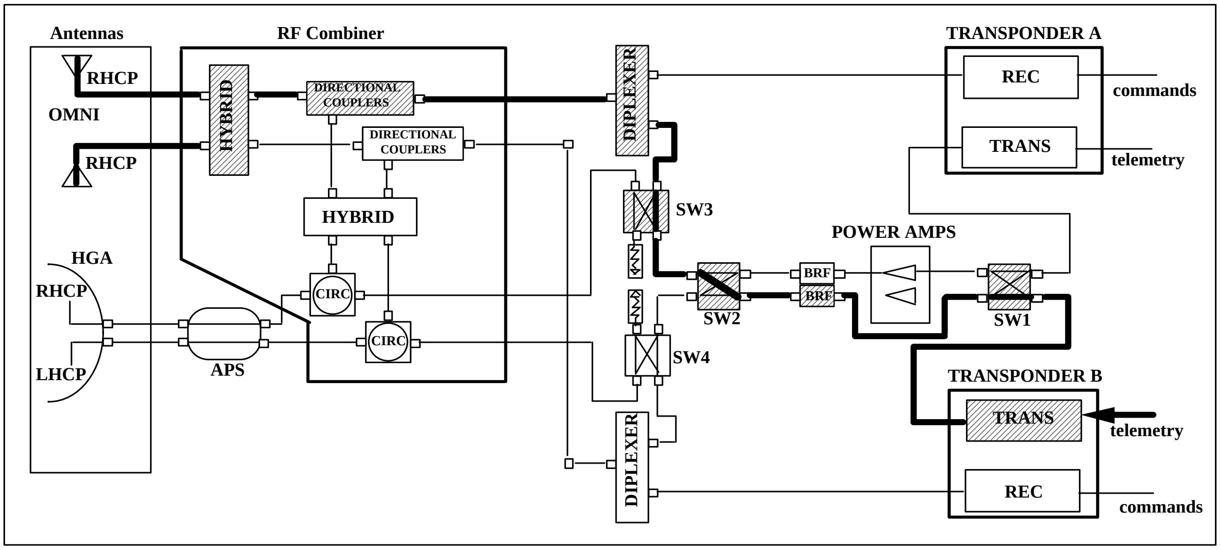


Figure 4.5-7 OMNI Transmit Path

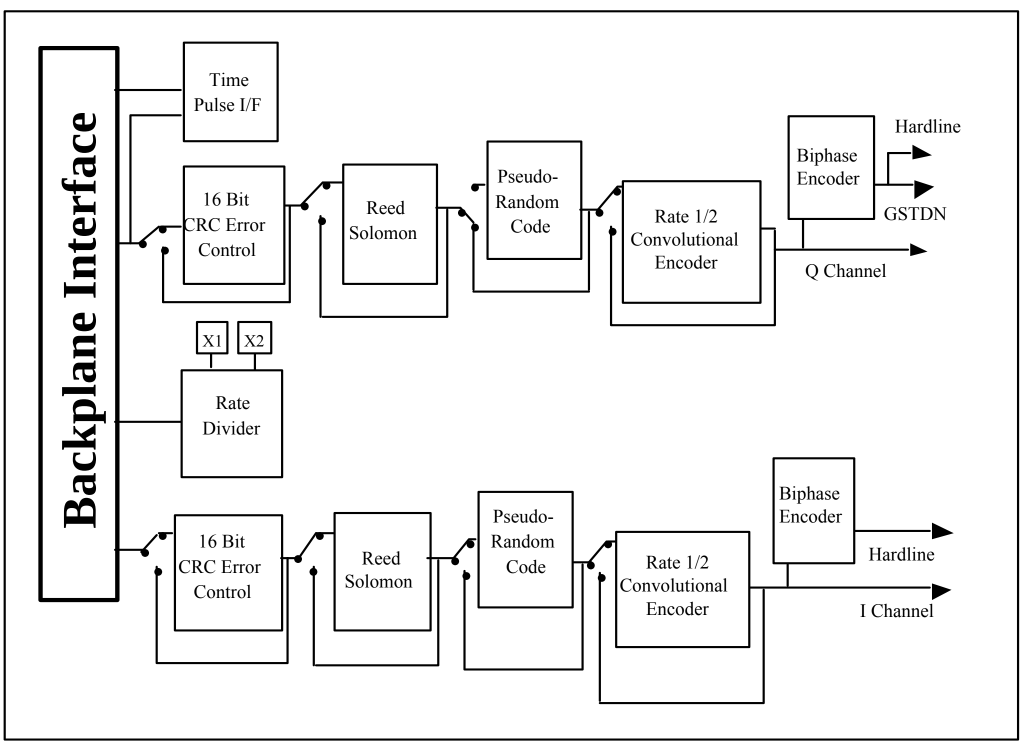


Figure 4.5-8 On-Board Encoding Schemes (Nominal Configuration)

For the coherent mode (DG1 Mode 1 or DG1 Mode 3), the long PN code is stripped from the forward carrier, as mentioned in section 4.5.3.1.1, and turned around on the return link carrier at a frequency 240/221 times the forward carrier frequency. This mode provides two-way range and Doppler measurements.

For the non-coherent mode, the I- and Q-channel codes are short Gold codes. The operation is exactly the same as that of the coherent mode, except that the return link carrier frequency is generated on board the spacecraft by the TCXO and there is no code synchronization to the forward link range channel PN code epoch. However, the frequency should be approximately 240/221 of the receiver frequency. The non-coherent mode will be used to obtain TCXO Doppler tracking data. The Flight Dynamics Facility processes the one-way tracking data to provide measurement of the transponder center frequency.

The return link signal is transmitted at 2255.50 MHz to the TDRS using the HGA. Real-time data will be downlinked at a rate of 32 Kbps on the TDRS I-Channel during TDRS SSA events. Solid State Recorder playback data will be played back on the Q-Channel at a rate of 2.048 Mbps. Specific return link rates are selectable from the FDS rate combination table. Table 4.5-3 shows the return link data rates and the associated spacecraft encoding configurations. In TDRS, the signal is coherently upconverted to Ku-Band and transmitted to the TDRSS ground terminal.

Data Rates I	Q	Real-Time Channel	Encoding Configuration
1 Kbps *	1 Kbps	Q	R-S Disabled & PN, CRC Enabled (I and Q-Channel)
1 Kbps *	1.5 Kbps	Q	R-S Disabled & PN, CRC Enabled (I and Q-Channel)
32 Kbps	128 Kbps	I	R-S, PN, & CRC Enabled (I and Q-Channel)
32 Kbps	2.048 Mbps	I	R-S, CRC Enabled (I and Q-Channel) & PN Disabled (Q-Channel)
N/A	1 Kbps	Q	R-S & PN Disabled, CRC Enabled
N/A	1.024 Mbps	Q	R-S & PN Disabled, CRC Enabled

* Fill data
N/A There will be one data downlink when in the GSTDN mode

Table 4.5-3 Selectable Downlink Configuration Rates

At the TDRSS ground terminal, the PN receiver acquires, tracks, and despreads the received signal. The I- and Q-Channels are then forwarded to the Binary PSK (BPSK) demodulators. The BPSK demodulates the transmit carrier, and provides the demodulated baseband signals (I and Q) to separate bit synchronizers. The recovered carrier is provided to the Doppler extractor for two-way or one-way Doppler measurements. The bit synchronizers recover symbol clock and provide the encoded symbols to Viterbi decoders. The NRZ-L output of the Viterbi decoder is provided to the TRMM MOC.

4.5.3.2.2 GN/DSN Return Link Operations

The S-Band downlink will be used to transmit data from TRMM to the GN/DSN with only one downlink supported at any given time. GN/DSN supports will be nominally be scheduled in the non-coherent mode; however, tracking service data will not be provided.

Merged recorder playback and real-time housekeeping data will be downlinked at 1.024 Mbps or only housekeeping data at 1 Kbps. The data will be converted from NRZ-L format to Biphase-L format. The Biphase-L will be used to phase modulate (PM) the transmitter in the transponder. The nominal frequency will be 2255.50 MHz.

The GN/DSN ground system will employ a PM receiver which will demodulate the transmitted signal. The demodulator will be processed by a bit synchronizer to provide telemetry data at the correct bit rate in NRZ-L format.

4.5.3.3 Transponder Center Frequency Maintenance

The TRMM second generation TDRSS user transponders include an internal TCXO which provides the frequency reference for both transmitter and receiver. The design includes the capability to command a Center Frequency Offset (CFO) of ± 80 KHz. During normal operations, the center frequency of each transponder will be determined from an evaluation of one-way tracking data acquired during non-coherent TDRS events. Such an evaluation will be performed on a weekly basis by the Flight Dynamics Facility (FDF). The results of the frequency evaluation will be provided electronically to the FOT in the Local Oscillator Frequency (LOF) report. The transponder Static Phase Error (SPE) telemetry (downlinked in the real-time data stream), can also be used to determine the TCXO drift.

Using the LOF, the FOT will trend the frequency drift for both transponder TCXOs for the life of the mission. The SN requires that the user transmit frequency be maintained to within ± 1500 Hz of the reference center frequency of 2255.50 MHz, as specified in the NCC configuration codes. The current FOT approach is to command center frequency offset adjustments whenever the frequency drift approaches ± 700 Hz from the center frequency reference.

4.5.4 RF Switch Operations

The Communications Subsystem includes four commandable RF switches to control the antenna configuration to the transponders. These RF switches protect against a single point failure by providing cross-strapping capability within the transmit paths. The four switches shown in Figure 4.5-1 are denoted as SW1, SW2, SW3, and SW4. Switch 1 (SW1) is used for the selection of the Power Amplifier (PA). This switch was originally designed to allow either PA to be configured to either transponder unit. **Note: Switch 1 is no longer connected to PA2 (a Bypass has been installed). If Switch 1 is configured to use PA2, no PA will be used.** Switch 2 (SW2) is used to route the signal from the selected PA and Band Reject Filter (BRF) to either Switch 3 (SW3) or Switch 4 (SW4). SW2, along with SW3 and SW4 also provide the command selectability between the HGAs right-hand circular (RHCP) and left-hand circular polarization (LHCP). SW3 and SW4 are also used to control the route of the return link signal through either of the Omni antenna or HGA. There are no RF switches in the Receive path of the transponder.

4.6 THERMAL SUBSYSTEM

The Thermal subsystem controls the thermal interfaces of the observatory. The Thermal subsystem will provide conductive and radiative interfaces for the TRMM instruments. Acceptable temperatures will be maintained in all mission modes. In addition, the Thermal subsystem assures that all thermal interfaces with the launch vehicle are satisfied. The Thermal subsystem consists of the following components:

a. Heater elements/thermostats
b. Heat pipes
c. Louvers
d. Temperature sensors
e. Solid State Temperature Controllers
f. Thermal blankets and blanket support hardware
g. Thermal coatings and miscellaneous materials

Heater elements, including thermostats, are used to maintain certain components above their minimum temperature limits. Heaters are used to maintain setpoint temperature control as needed and are controlled by series redundant thermostats. In addition, all heater circuits are redundant and are attached to selected surfaces on the spacecraft components.

Thermostats are ON/OFF control types and are spring actuated. Two units, connected in series, provide redundancy to control each heater circuit. Thermistors are temperature sensors which monitor temperatures at various locations on the spacecraft. These units are read-only telemetry points.

Heat pipes are used where the necessary thermal performance can not be provided by conductive and radiative interfaces. On the TRMM spacecraft, heat pipes are used to transport communication equipment energy to the "cold" side of the spacecraft (+Y side). The heatpipes are located on the +X bulkhead of the TRMM spacecraft, under the electronics boxes, in order to maximize the heat that is transferred. Heat pipes are fully redundant.

Louvers control spacecraft component temperatures in a localized area by modulating heat rejection in response to the temperature in that area. Louver blades are activated by attached actuators and support a 10° C differential between opening/closing temperatures. They are mounted on the spacecraft body on the +YZ, -YZ, and +Y panels of the Lower Bus Structure as shown in Figure 4.6-1.

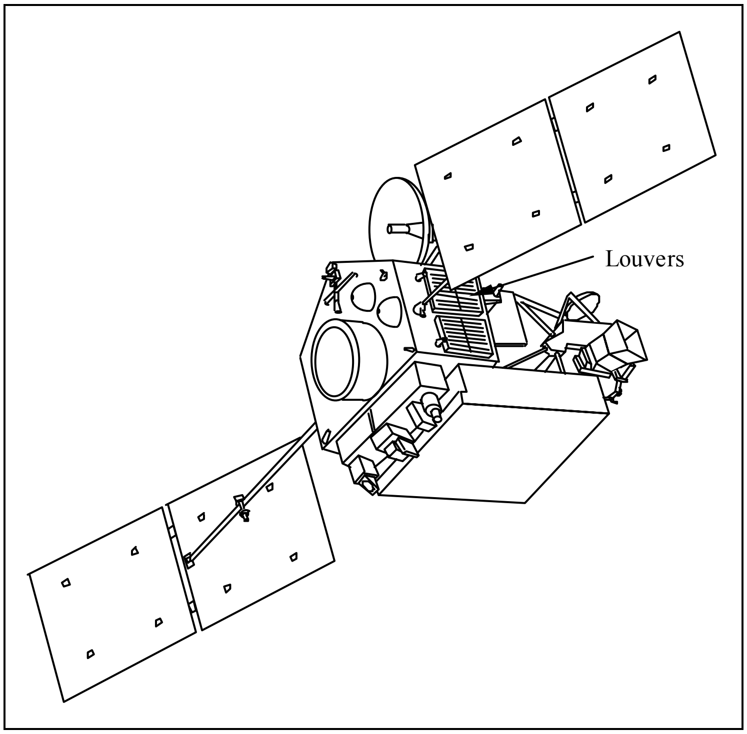


Figure 4.6-1 Louvers

Solid State Temperature Controllers (SSTCs) are also used to maintain a component above its minimum operating temperature requirements and to meet component temperature stability requirements. On TRMM, SSTCs are used to precisely maintain the temperature of the IRU and fuel lines.

Temperature sensors will be provided to monitor temperatures at significant locations on the spacecraft. These sensors will contain elements whose electrical resistance varies as a function of temperature. Passive analog telemetry capabilities will be used to monitor the resistance. Resistance versus temperature curves will be provided for all sensors.

Thermal blankets and thermal coatings will be used to regulate radiant heat transfer between the spacecraft and the space environment. All external radiators are coated with an AO resistant, UV stabile white paint to minimize the incident solar energy absorbed by the observatory. Except for the radiator, the entire observatory is covered with MLI blankets. High emittance coatings are used to maximize the radiation heat transfer on the interior of the Lower Bus Structure (LBS) and components, except on propulsion components. The propellant tanks and fuel lines will have a low emittance coating to minimize radiation heat loss. There are no unique telemetry points related to these items. They have an overall effect to the general thermal balance of the spacecraft.

4.6.1 Thermal Subsystem Operations

With regard to FOT activity, the Thermal subsystem is rather passive. The only thermal requirement is to keep the Sun on the -Y side of the spacecraft. The anti-Sun side is used to dissipate heat from the electronics that do not have a dedicated radiator, such as the communications equipment and the SDS.

Temperature mnemonics for all subsystems and instruments will be monitored on a routine basis during all real-time supports. Heaters on the non-essential bus may be commanded ON or OFF as a situation warrants. These command operations occur as the subsystem engineers direct or via pre-arranged guidelines. While heater commands could execute out of onboard memory, they will normally be transmitted as part of a real-time support. In any case, requests for thermal commanding of any sort are expected to be rare.

Command transmission will be routed through the prime FDS Uplink Card which then passes it to the appropriate PSDU box. From there, the correct heater relay is pulsed either opened or closed. Various heaters are located in the SPSDU although primary power distribution is supplied to instrument survival heaters by the IPSDU.

The Table 4.6-1 presents the operating and survival temperature limits for major spacecraft components. In general, these component temperatures are qualified during I&T testing to ± 10 degrees of the operating limits.

4.6.2 Performance Monitoring

Temperature mnemonics are defined and Packetized to be consistent for data sampling interval. In this way, telemetry points with similar downlink sample rates are available in the same telemetry packet. Visual and configuration monitor functions will be provided in the MOC to repeatedly check safe operational status. Out-of-limits or other anomalous conditions are flagged for prompt response. Off-line trending of thermal mnemonics is performed as necessary.

COMPONENT	OPERATING LIMIT °C	SURVIVAL LIMIT °C (Non-Operating)
ACS		
ACE	0 to 40	-25 to 60
CSS	-125 to 65	-135 to 90
ESA	-5 to 25	-25 to 60
EVD	0 to 40	-25 to 60
DSS	-30 to 60	-135 to 90
IRU	12 to 40	-25 to 60
MTB	-30 to 60	-45 to 70
RWA	0 to 40	-25 to 60
TAM	-30 to 65	-50 to 85
C&DH		
Frequency Standard	0 to 40	-25 to 60
SDS	0 to 40	-25 to 60
Star Couplers	0 to 40	-25 to 60
Communications		
Band Reject Filter	-10 to 45	-25 to 60
Omni	-90 to 80	-150 to 100
PA	-10 to 50	-25 to 60
RF Combiner	-10 to 50	-10 to 50
XPNDRs	-10 to 45	-25 to 60
Deployables		
GSACE	0 to 40	-25 to 60
HGA Actuator	0 to 40	-20 to 50
Damper	30 to 50	-20 to 70
SADA Actuator	0 to 40	-20 to 50
Electrical		
IPSDU/SPSDU	0 to 40	-25 to 60
Power		
Batteries	0 to 20	-20 to 35
PBIU	0 to 40	-25 to 60
PSIB	0 to 40	-25 to 60
Solar Arrays	-85 to 85	-100 to 100
SPRU	0 to 40	-25 to 50
Propulsion		
FDM	8 to 40	8 to 50
Fuel Lines	8 to 65	8 to 70
PCM	8 to 40	8 to 50
Pressurant Tank	-40 to 40	-50 to 50
Propellant Tanks	8 to 40	8 to 50

Table 4.6-1 Temperature Limits

COMPONENT	OPERATING LIMIT °C	SURVIVAL LIMIT °C (Non-Operating)
Instruments @ S/C I/F		
CERES	-10 to 40	-10 to 50
LIS	-10 to 40	-10 to 50
PR	-20 to 50	-30 to 60
TMI	-25 to 40	-25 to 50
VIRS	-25 to 40	-25 to 50

Table 4.6-1 Temperature Limits (continued)

4.7 REACTION CONTROL SUBSYSTEM

The Reaction Control Subsystem (RCS) provides the necessary impulses required to perform initial descent maneuver, orbit maintenance (maintain orbit altitude within ± 1.25 km), and a safe end-of-life ocean disposal. The RCS also provides a backup momentum wheel unloading and yaw maneuver capability. The RCS may also be used to null any launch vehicle induced tip-off rates, if the rates exceed the capacity of the reaction wheels and magnetic torquer bars. The RCS will also provide for the initial transfer from orbit insertion altitude (380 ±10 km) down to approximately 360 km. Components of the RCS include a pressure tank, two propulsion tanks, the Propellant Tank Module (PTM), Propellant Control Module (PCM), Fill and Drain Module (FDM), twelve Rocket Engine Modules (REMs), pressurant transducers, filters, tubing, and five isolation valves. Figure 4.7-1 provides a graphical illustration of the RCS. RCS components have redundant thermostatically controlled heaters to prevent the hydrazine from freezing.

The RCS operates in a regulated pressure or blowdown mode. Thrust is provided by twelve REMs, otherwise known as thrusters, by the decomposition of hydrazine (N_2H_4).

4.7.1 Tanks (Propellant and Pressure)

The TRMM RCS includes two propellant tanks and a pressurant tank. The three tanks will provide the necessary fuel and pressure required to perform the orbit maintenance maneuvers of the mission.

4.7.1.1 Propellant Tanks

The TRMM design includes two positive expulsion, elastomeric diaphragm propellant tanks manufactured by Pressure Systems Incorporated. These tanks were titanium construction flight qualified on the TDRS program. The PTM consists of the two propellant tanks, heaters, thermostats and thermistors, and associated tubing. As the tanks drain, the center of gravity travels along the X-axis of the spacecraft to minimize disturbance to ACS control.

The aft Omni antenna is mounted to the aft propellant tank.

4.7.1.2 Pressurant Tank

The pressurant tank, manufactured by Pressure Systems Incorporated, provides a regulated supply of pressure to the propellant tank modules. The pressurant tank is an all-titanium design, and contains Gaseous Nitrogen (GN_2).

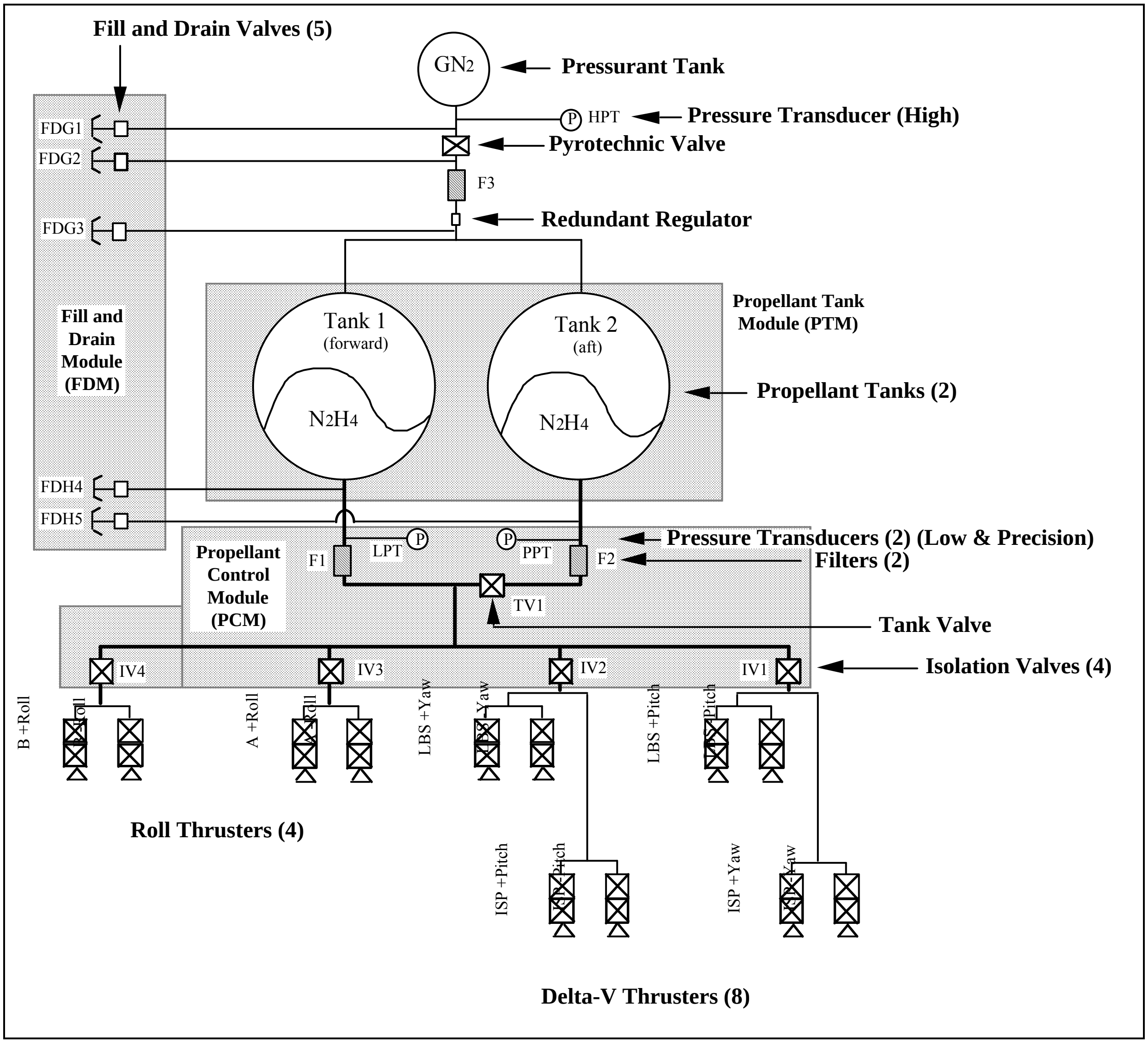


Figure 4.7-1 TRMM RCS Diagram

The RCS will operate in a pressure regulated mode (nearly constant pressure of 167 ± 12.5 psi) for the majority of the mission. Towards the end of the mission, as the tank begins to empty, the gas will naturally begin to expand causing a decrease in the pressure.

4.7.2 Propellant Control Module

The Propellant Control Module (PCM) controls propellant flow paths, filters propellant, and provides pressure related telemetry. The PCM is composed of five magnetically latched, two-position isolation valves, two filters, and two pressure transducers (one low and one precision).

4.7.2.1 Isolation Valves

The RCS is equipped with five isolation valves (a Tank Isolation Valve and four Thruster Isolation valves). The tank isolation valve prevents the aft propellant tank from being overfilled during propellant loading and launch. The tank isolation valve, CLOSED for launch, will be commanded OPEN as a part of the inital in-orbit check-out, possibly in conjunction with the firing of the pyrotechnic valve to pressurize the RCS. Once the tank isolation valve is OPEN, it should not be commanded to CLOSE again for the duration of the mission (barring any unforeseen anomalous conditions to the RCS). The four thruster isolation valves distribute propellant to the groups of REMs and remain OPEN during launch and in all nominal mission operations modes. The thruster valves are only CLOSED when a thruster leaks uncontrollably, or during any mode in which the ACS is unable to maintain the required observatory attitude rates. If during thruster operations (Delta-V or Delta-H maneuvers) it becomes necessary to shut down the system, the sequence for commanded shutdown is to CLOSE the thruster valves before closing the isolation valves. Not doing so can increase the possibility of extremely high pressure surges in the lines occurring when the isolation valves are opened.

Table 4.7-1 reflects the operational status of the tank and thruster isolation valves.

Valve	Launch Configuration	Mission Operations
Tank Isolation Valve	Closed	Open
Thruster Isolation Valve	Open	Open

Table 4.7-1 Operational Valve Configurations

4.7.3 Rocket Engine Modules

The TRMM RCS consists of twelve REMs that are located about the spacecraft as shown in Figure 4.7-2. Four thrusters are mounted on the Fore (+X) end of the spacecraft, four thrusters are mounted to the Aft (-X) end of TRMM, and four roll control thrusters are located on the sides of the spacecraft (2 +Y and 2 -Y).

The REMs provide the necessary impulse required to maintain the orbit of the mission. Operationally, catalyst bed heaters will be powered ON prior to any firing of the REMs. A minimum temperature of 90° F must be obtained prior to firing the REMs. However, the REM design allows a number of Cold firings (firing thrusters at a temperature lower than 90° F) during the course of the mission, if necessary.

There is no direct interaction between the 1773 command bus and the RCS. All thruster and valve actuation occurs through the ACE/EVD interface. Figure 4.7-3 provides a graphical illustration of the thruster firing command path. Likewise, all pressure transducer and catalyst bed heater actuation occurs via the PSDU. Additional details with respect to the ACS-to-ACE-to-EVD interface are described in section 4.2

4.7.4 On-orbit Operations

The RCS requirements are based on performing nominal mission operations throughout the mission lifetime. On-orbit capabilities of the RCS include Orbit adjust maneuvers and EOL ocean disposal. Additional capabilities of the RCS include Backup yaw maneuver and momentum wheel unloading. Operationally, there are no normal operations commanding (ground) of the RCS, as Delta-V and Delta-H maneuvers are controlled via the ACS-to-ACE-to-EVD interface.

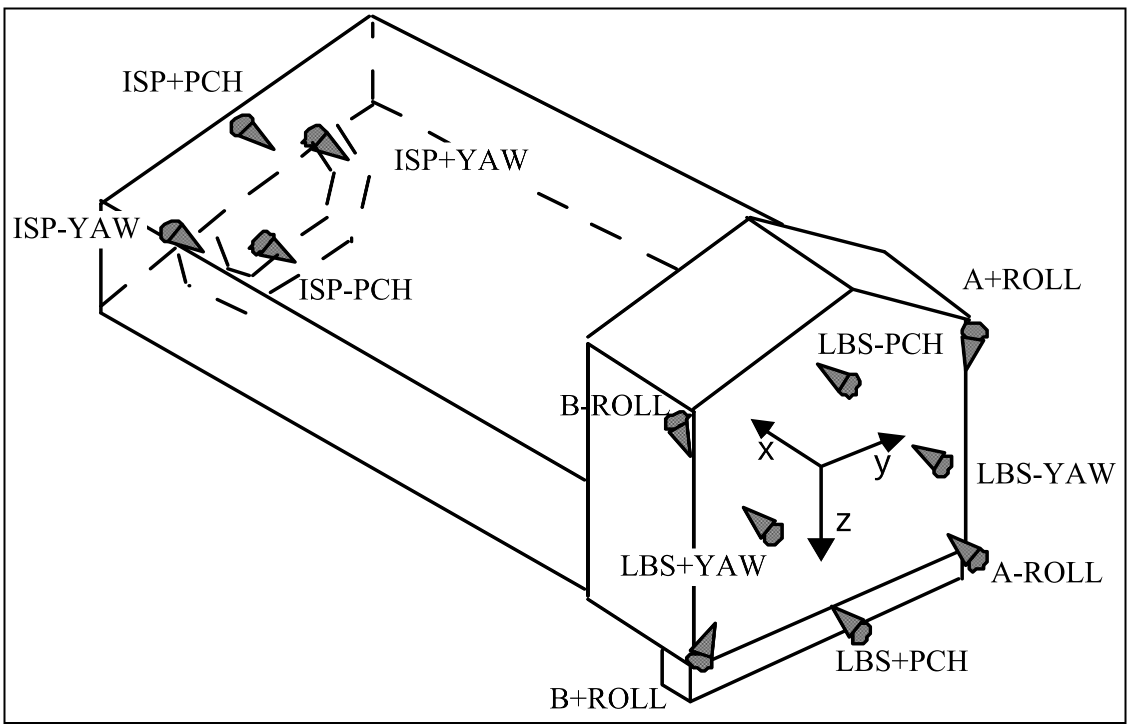


Figure 4.7-2 Thruster Configuration

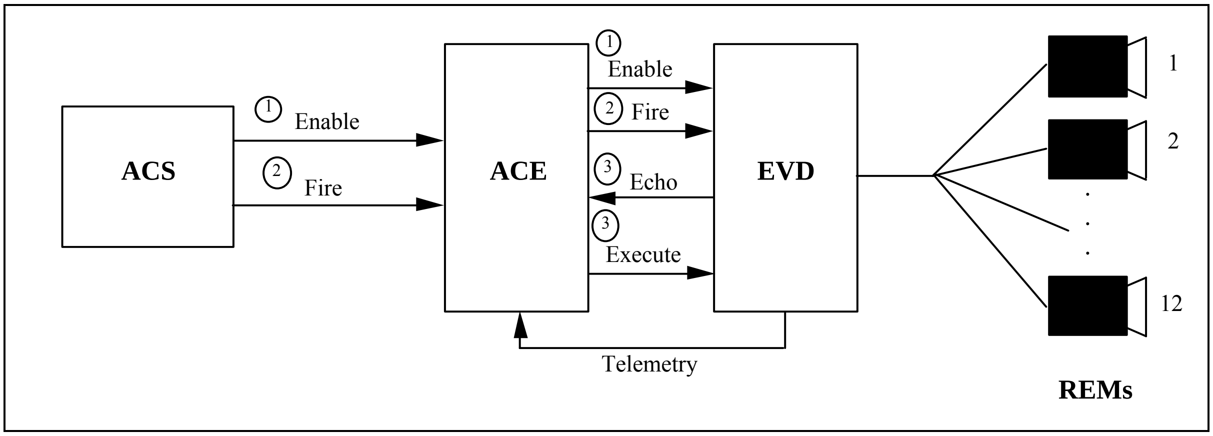


Figure 4.7-3 REM Firing Command Path

4.7.5 EOL Ocean Disposal

The EOL ocean disposal of TRMM shall be in accordance with the requirements stated in the NASA Handbook (NHB 1700.1). The spacecraft altitude will be allowed to decay to 200 km. Magnetic torquer bars will be used to unload reaction wheel momentum from the nominal mission altitude, down to 270 km. The RCS will provide momentum unloading of the reaction wheels during the remainder of the descent to 200 km. The RCS shall support a final disposal maneuver to place the spacecraft into an elliptical orbit with a target perigee of 50 km.

4.8 DEPLOYABLES

The Deployables subsystem provides the capability to deploy and control the SA and HGA appendages. The components of the Deployables subsystem are the High Gain Antenna System (HGAS), Gimbal Solar Array Control Electronics (GSACE), and the Solar Array Deployment and Drive System (SADDS).

4.8.1 High Gain Antenna System

The HGAS provides the capability to communicate with the ground, via TDRSS. It consists of a High Gain Antenna, a Gimbal System, the GSACE, and a High Gain Antenna Deployment System (HGADS). Figure 4.8-1 provides a functional block diagram of the HGAS.

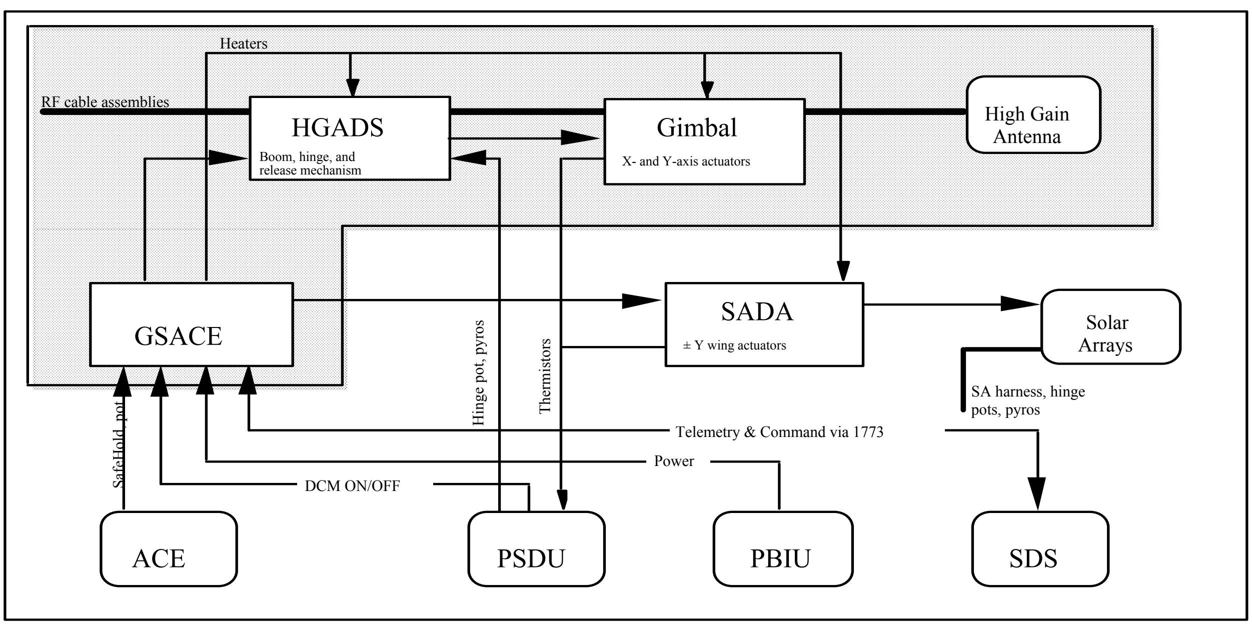


Figure 4.8-1 HGAS Functional Block Diagram

4.8.1.1 High Gain Antenna

The HGA is a parabolic dish with a diameter of 1.323 meters and a height of 0.587 meters. The HGA is supported by a small pedestal. The HGA is the primary component of the Communications subsystem, used to transmit the high data rate of TRMM to the ground. Stowed for launch, the HGA will be deployed, via the HGADS, by the SPSDU sequencer after separation from the H-II launch vehicle. On-orbit operations of the HGA are discussed in section 4.5.

4.8.1.2 High Gain Antenna Deployment System

The TRMM possesses a HGADS to support and deploy the HGA, the Gimbal System, and the harness. The HGADS consists of a single aluminum boom with redundant pyrotechnic pin pullers, dampers, hinges, and sensors.

At launch, the HGADS is folded and stowed, including the gimbals and antenna, such that they fit within the H-II fairing. After separation from the H-II launch vehicle, deployment of the HGADS is initiated by sequencer commands activating the pyro pin pullers after deployment of the Solar Arrays. In addition to deployment of the HGAS, the HGADS provides the structural support for the HGA during the mission.

4.8.1.3 Gimbal

The HGA has a gimbal system that consists of two gimbal motors, called the Outer and Inner gimbal. The angle that the Outer gimbal is at is called the α angle, while the Inner Gimbal angle is called the β angle.

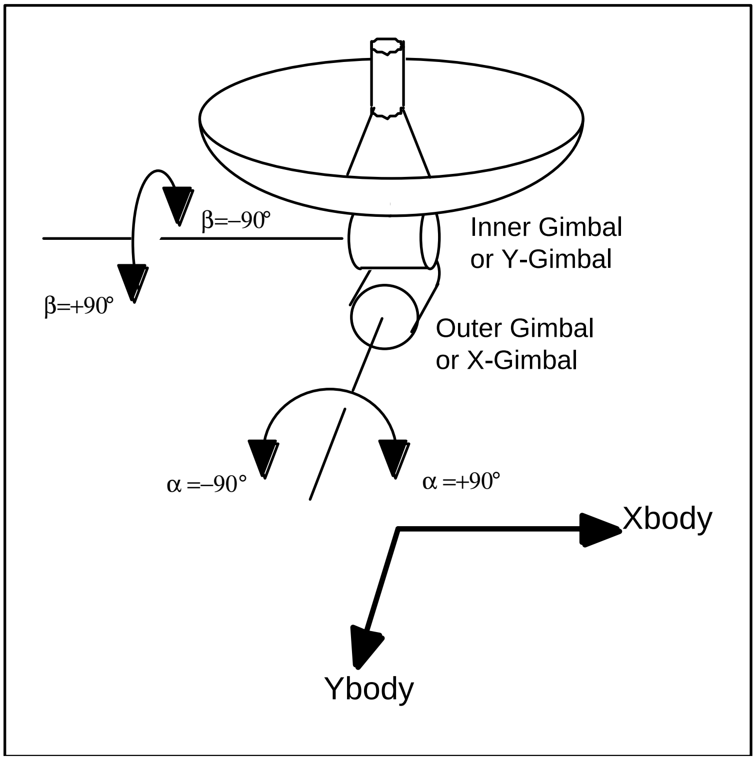


Figure 4.8-2 HGA Gimbal Orientation

Range Of Movement

Each gimbal is physically capable of rotating +/- 99°. At this point there is a hardware stop. It will not damage the antenna to come up against the hardware stop.

In addition, there are a number of software stops built into the system. When the ACS is controlling the HGA, the ACS software will only command them to +/- 90.2°. Furthermore, the GSACE electronics will check commands sent to it. When GSACE is in Closed Loop Mode, (this mode is described below), then it will only allow commands of up to +/- 98°.

Commands from the ground can be sent to the GSACE to control the pointing as well. The GSACE can be put in 'Open Loop Mode'(also described later), when commands are sent from the ground. When the GSACE is in 'open loop mode', the range of motion is limited only by the hardware stops on the HGA. Commanding the gimbals up against the hard stops will not damage them.

Keyhole

The HGA cannot freely move through the entire hemisphere. When the Inner gimbal is +/-90°, the antenna will point along either the + or - Y body axis, regardless of what angle the inner gimbal is commanded to. Another way of looking at this is to think about tracking an object that is moving along the edge of the hemisphere. When it passes the Y axis, the Outer Gimbal would instantly have to change from -90° to +90° to keep tracking it. It cannot do so. For this reason, a 7.5° exclusion zone is defined around both the +Y and -Y body axes. The antenna cannot track objects in this zone.

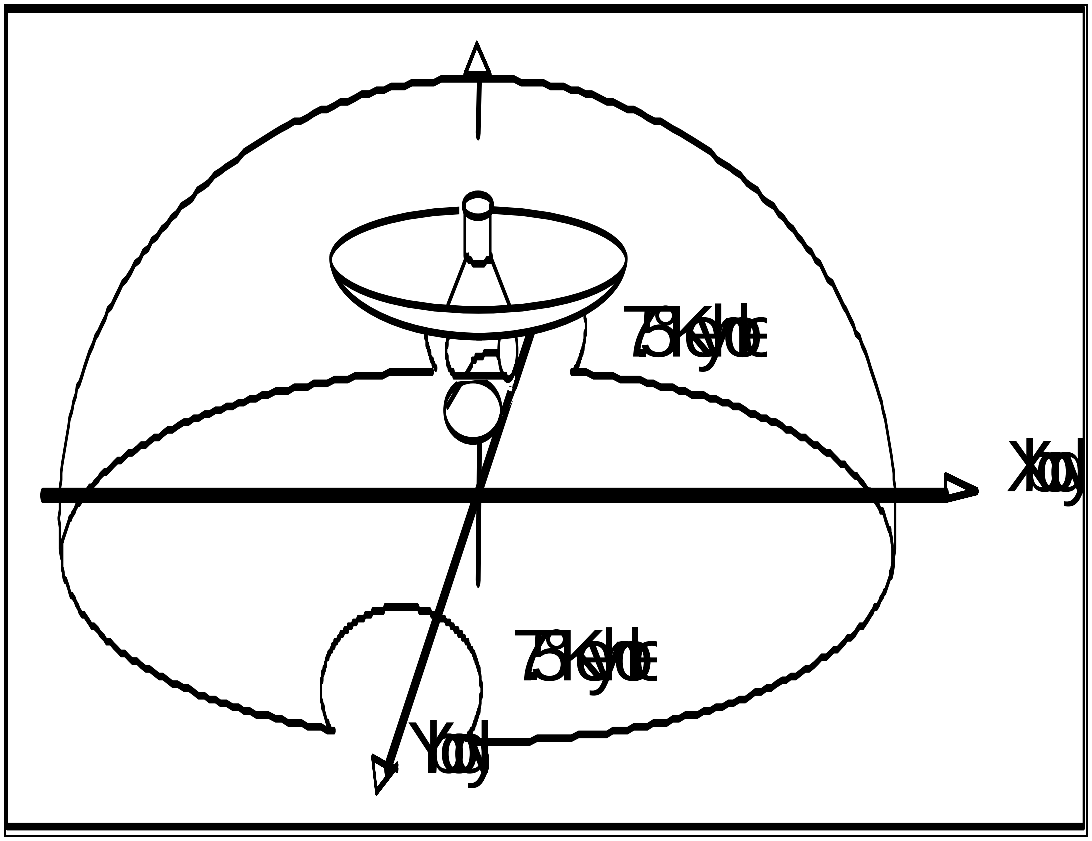


Figure 4.8-3 HGA Range of Motion and Keyhole

4.8.2 Solar Array Deployment and Drive System

The SADDS consists of two independent deployable Solar Array wings. Each wing consists of the following major elements:

a. Solar panel assembly
b. Deployment boom
c. Solar Array Drive Assembly (SADA)
d. Restraint mechanisms

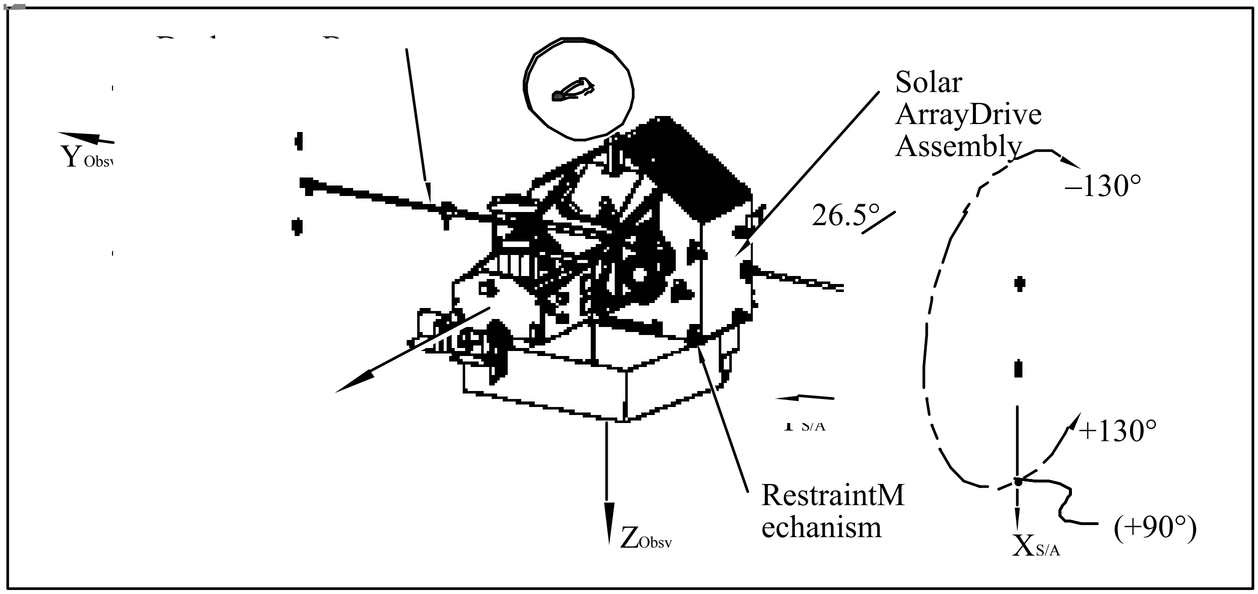


Figure 4.8-4 SADDS Components

The SADDS provides the mechanical interface to the SA components and CSSs. Figure 4.8-5 provides a functional block diagram of the SADDS and its interfaces with the Power, Electrical, and ACS subsystems. The following sections provide a description of the components of the SADDS.

4.8.2.1 Solar Panel Assembly

Each Solar Array consists of two solar panels (inboard and outboard panel) connected by two panel hinges. Each panel contains the solar array cells used by the Power subsystem. Also, two Course Sun Sensors are mounted to each of the outboard solar panels. In addition, the hinges that connect the inboard and the outboard panels include a potentiometer whose angular position will be reported in telemetry.

4.8.2.2 Deployment Boom

The Deployment Boom provides the connection between the solar panel assembly and the SADA. It also provides the energy required for deployment of the solar panel assemblies. Potentiometers provide position data of the hinge positions during deployment.

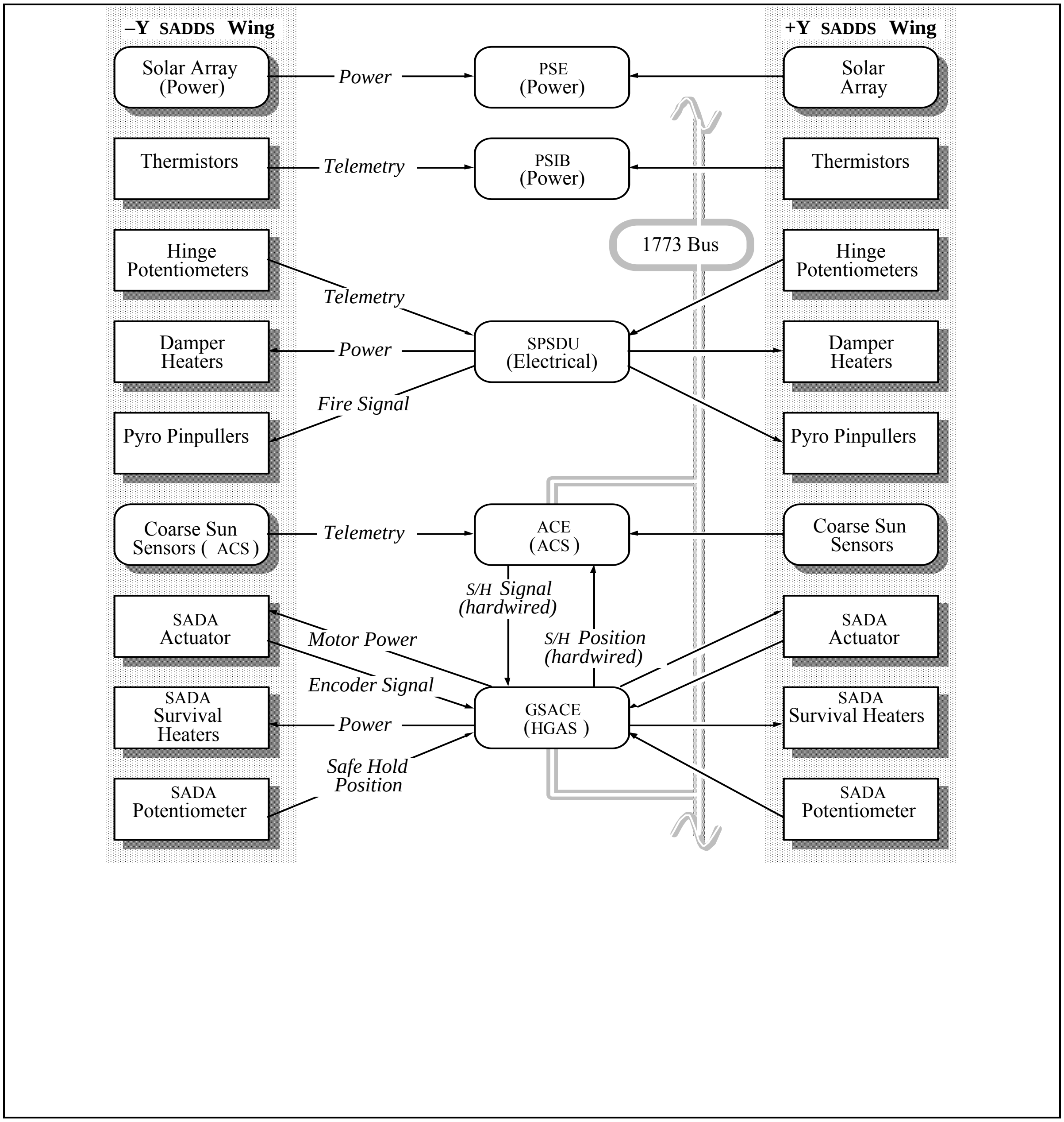


Figure 4.8-5 SADDS Functional Block Diagram

4.8.2.3 Solar Array Drive Assembly

The SADAs act as conduits of SA power and telemetry. They also provide the ability to rotate the SA booms around the Y axis of the spacecraft, so that the arrays may be oriented to collect the most sunlight. There are two SADAs, one for the SA boom pointed along the +Y axis, and the second for the SA boom pointed along the -Y axis of the spacecraft.

Range of Motion
Each SADA motor is physically capable of rotating ± 174° about the Y axis. At this point there is a hardware stop. It will not harm the motor to come up against the hardware stop.

In addition there are a number of software stops built into the system. When the ACS is controlling a SADA, the ACS software will only command them to ±130°. Furthermore, the GSACE electronics will check commands sent to it. When the GSACE is in Closed Loop Mode, (this mode is described below) then it will only allow commands of up to ±130.8°.

One particular item should be emphasized. This motor is not designed to rotate continuously. It can rotate forward as far as 130°, and then slowly rotate to -130°. At that point it stops rotating. If the Sun were to re-appear at +130° again, the SA must wind back 260° to get back to +130°. It cannot simply keep moving through the 100° separating -130° and +130°. This however will not affect operations. The plan is to feather the SA while TRMM is in the shadow of the Earth. This means that the SA will always be brought back to zero between daylight portions of the orbit.

Commands from the ground can be sent to the GSACE to control the pointing as well. The GSACe can be put in 'Open Loop Mode' (also described later), when commands are sent from the ground. When the GSACE is in this mode, the range of motion is limited only by the hardware stops on the SADA. Commanding the SADA up against the hard stops will not damage it.

4.8.2.4 SADDS Deployment

Restraint mechanisms hold the Solar array wings and deployment boom in the stowed position during launch and ascent of the spacecraft. Upon spacecraft separation from the H-II launch vehicle, pairs of Pyros are fired to release the wings.

When the spacecraft separates from the H-II, the Solar Array wings are released by sequencer commands activating the pyro pin pullers. Deployment to the fully opened position is achieved through the use of springs, however rotary viscous dampers within the hinges will limit the deployment rate. The panels will lock into place upon reaching the 180° end-of travel. SA panel position (within 5°) telemetry during deployment is provided by potentiometers.

It should be noticed in figure 4.8-5 that pyro control as well as potentiometer telemetry is provided via the SPSDU.

4.8.3 Gimbal Solar Array Control Electronics

The GSACE provides an interface between the ACS processor and the Deployables. The best description available of it's operation is provided in the TRMM ACS-GSACE ICD [TRMM-712-157]. Ordinarily, control of the HGA and SA is handled by the ACS, rather than the ground. Command packets are formulated by the ACS and addressed to the GSACE over the 1773 Bus.

It is possible to directly command either the SA or GSACE, by formulating packets using the same addressing method. The packet format is described in detail in the ACS-GSACE ICD.

4.8.3.1 GSACE Control Modes

The GSACE has two control modes. They are the Open Loop and Closed Loop Control Modes.

Closed Loop Mode
When the GSACE is in the Closed Loop mode, it keeps track of the current position of each motor. When a command arrives directing the motor to a new position, the GSACE compares the new and old position, and deduces which way the motor must turn to achieve the new position. Furthermore the GSACE monitors the commands. It will not allow the motors to be driven beyond +/- 98°. Nominally, the Closed Loop Mode will be utilized.

Open Loop Mode
In Open Loop Mode the GSACE does not employ knowledge of the current position of the antenna. Instead, commands sent to the GSACE must specify the direction that a given motor must turn as well as the number of pulses to be sent to the motor. Each pulse will turn the motor .0075°.

When the ACS sends positioning commands to the GSACE, it is expecting the GSACE to be in Closed Loop Mode.

4.8.3.2 GSACE Monitor of the ACE Safehold Pulse

The GSACE can be commanded to monitor any combination of ACE-A and ACE-B SafeHold pulses. When both ACEs are functioning properly, the GSACE should be set to monitor both ACE-A and ACE-B. If a Safehold pulse comes from either, the GSACE will go into SafeHold. One can also command the GSACE to monitor ACE-A only or ACE-B only (e.g. if ACE-A had failed). In that case, during recovery from that failure, one would configure the GSACE to only monitor the ACE-B pulse in the future.

There is one last option provided. It is to monitor neither ACE-A nor ACE-B. This risky choice should not be used. It is left over from XTE requirements.

4.8.3.3 GSACE and the Gimbal Velocity

The HGA Gimbal motors are capable of moving as fast as 90 degrees per minute. While tracking TDRS, the motors will not have to go nearly that fast. Typical speeds while tracking will be about 4° per minute. The 90 degrees per minute limitation affects HGA commanding by the ground as described earlier.

The Velocity Command is used to specify the rate that a given motor will step at. The command has a 16 bit field, so the possible range of counts is from 0 to 65535 counts. This entire range of counts are not usable however. To see this consider the equation that relates counts to desired degrees per minute. This equation is:

$$(\text{desired degrees per minute}) = (0.45 * 15625) / (\text{Number of counts})$$

If you put in 1 count, the answer is 7031 degrees per minute. The motor however is only designed for 90 degrees per minute. 90 degrees per minute equals 78 counts. So the valid range is 78 counts to 65535 counts. If you stay in this count range, the motor speed can be from 0.1073 degrees per minute to 90.125 degrees per minute.

The motor was actually tested to 135 deg/min., but this number cannot be guaranteed. If for example the motor temperature is not just right, you'll get skipping. For example if 5 counts were commanded, the HGA might only move 4 counts and shake a bit as well. In general, if you try to command too high a rate, the motor will not be damaged, but it may not do what you expect. For just a slightly higher rate, it will skip. If a very high velocity is commanded, the motor will not move at all, instead it will just shake.